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PROCEEDINGS OF THE ANNUAL MECHANICS OF COMPOSITES REVIEW (13^{TH})

Sponsored by:

Air Force Wright Aeronautical Laboratories
Materials Laboratory

APRIL 1997

FINAL REPORT FOR PERIOD 2-3 NOVEMBER 1988

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MATERIALS DIRECTORATE
WRIGHT LABORATORY
AIR FORCE MATERIEL COMMAND
WRIGHT-PATTERSON AFB OH 45433-7734

REPORT DOCUMENTATION PAGE			Form Approved OMB No. 0704-0188

1. AGENCY USE ONLY *(Leave blank)*	2. REPORT DATE	3. REPORT TYPE AND DATES COVERED
	April 1997	Final Report 2-3 NOVEMBER 1988

4. TITLE AND SUBTITLE

PROCEEDINGS OF THE ANNUAL MECHANICS OF COMPOSITES REVIEW (13th)

5. FUNDING NUMBERS

6. AUTHOR(S)

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

Air Force Materials Laboratory
Nonmetallic Materials Division
Wright-Patterson AFB OH 45433

8. PERFORMING ORGANIZATION REPORT NUMBER

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)

Materials Directorate
Wright Laboratory
Air Force Materiel Command
Wright-Patterson AFB Ohio 45433-7734
POC:Tammy Oaks, WL/MLBM, 937-255-3068

10. SPONSORING/MONITORING AGENCY REPORT NUMBER

WL-TR-97-4078

11. SUPPLEMENTARY NOTES

12a. DISTRIBUTION AVAILABILITY STATEMENT

APPROVED FOR PUBLIC RELEASE; DISTRIBUTION IS UNLIMITED

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*

This report contains the basic unedited vu-graphs of the presentations at the "Mechanics" of Composites Review" sponsored jointly by the Non-metallic Materials Division of the Air Force Materials Laboratory, the Structures Division of the Air Force Flight Dynamics Laboratory and the Directorate of Aerospace Sciences of the Air Force Office of Scientific Research. The presentations cover current in-house and contract programs under the sponsorship of these three organizations.

14. SUBJECT TERMS

epoxy-matrix composites; composite materials; resin matrix composites; composite bonded joints; fatigue of graphite/epoxy composites; fracture and fatigue of bi-materials

15. NUMBER OF PAGES

214

16. PRICE CODE

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
UNCLASSIFIED	UNCLASSIFIED	UNCLASSIFIED	SAR

AGENDA
MECHANICS OF COMPOSITES REVIEW
2-3 NOVEMBER 1988

WEDNESDAY, 02 NOVEMBER 1988

THURSDAY, 03 NOVEMBER 1988 PAGE

FOREWORD

This report contains the abstracts and viewgraphs of the presentations at the Thirteenth Annual Mechanics of Composites Review sponsored by the Materials Laboratory. Each was prepared by its presenter and is published here unedited. In addition, a listing of both the in-house and contractual activities of each participating organization is included.

The Mechanics of Composites Review is designed to present programs covering activities throughout the United States Air Force, Navy, NASA, and Army. Programs not covered in the present review are candidates for presentation at future Mechanics of Composites Reviews. The presentations cover both in-house and contractual programs under the sponsorship of the participating organizations.

Since this is a review of on-going programs, much of the information in this report has not been published as yet and is subject to change; but timely dissemination of the rapidly expanding technology of advanced composites is deemed highly desirable. Works in the area of Mechanics of Composites have long been typified by disciplined approaches. It is hoped that such a high standard of rigor is reflected in the majority, if not all, of the presentations in this report.

Feedback and open critique of the presentations and the review itself are most welcome as suggestions and recommendations from all participants will be considered in the planning of future reviews.

Deborah Mueller

DEBORAH C. MUELLER, Meeting Manager
Mechanics & Surface Interactions Branch
Nonmetallic Materials Division
Materials Laboratory

ACKNOWLEDGEMENT

We wish to express our appreciation to the authors for their contributions; to the focal points within the organizations for their efforts in supplying the program listings; and to Sally Lindsay, the Conference Secretary, for managing registration.

PROGRESS AND FUTURE CHALLENGES IN THE MECHANICS OF COMPOSITES

George K. Haritos
Air Force Office of Scientific Research
Bolling Air Force Base, DC 20332-6448

ABSTRACT

The invention of high-modulus boron and carbon fibers twenty-five years ago started a revolution in the design of structural materials whose potential impact is just beginning to emerge.

The Air Force has long recognized the potential of composite structural materials for aerospace applications. As early as 1964, General Bernard Schriever, addressing the NATO Defense College in Paris, called the development of boron fiber composites "...the most significant step forward in materials in ...3000 years..." (Ref 1, 2). Accordingly, the Air Force has supported and continues to agressively support research in composites conducted at Universities, Industry, and Government Laboratories.

Over the last two decades structural composites have gained considerable acceptance in a variety of industrial products. For example, their usage in military aircraft has increased from 2% in the Air Force's F-15, to 10% in the Navy's F-18, to 30% in the AV-8B Harrier (Ref 2). More significantly, projections place their usage at 50 to 60% in the Advanced Tactical Fighter (ATF) and Bomber (ATB), 70% in the V-22 tilt-rotor aircraft, and even 85 to 90% in the Army's experimental Light Helicopter (LHX) (Ref 3).

Before such projections become reality, however, the aerospace community must allay many concerns dealing with reliability, survivability, repairs, and life cycle costs of composite structures. In May 1987, for instance, during live-fire testing, a 30mm round blasted an 18 to 24-inch hole into the composite wing of the Harrier aircraft. Moreover, major structural damage occurred within an area 30 to 40% larger than the actual hole (Ref 3). Although this is an extreme case, it underlines the need for better fundamental understanding than presently available of how damage initiates, grows, and ultimately fails composite structure components.

At present, limitations in the understanding of the behavior of composite materials have restricted their role to being secondary to that of monolithic materials. However, their potential remains virtually boundless. As the 21st Century promises to take technology to an era of hypersonic flight, space-based operations, etc, structural composites seem poised to assume a dominant role in the development of future aerospace systems.

Projected requirements for emerging aerospace structures and engine, such as operating temperatures and temperature gradients, structural to gross weight ratios, engine thrust to weight ratios, etc, have, in many cases, virtually eliminated conventional monolithic materials as candidates for these applications. Instead, the focus of attention has shifted to such emerging material systems as ceramics, ceramic matrix composites, carbon/carbon composites, metal matrix composites, and a host of hybrid composites. Without exception, all of these emerging materials are expected to be highly anisotropic and inhomogeneous.

The Air Force is interested in sponsoring research aimed at developing analytical, experimental, and computational tools which will enable the identification, classification, and mathematical description of the deformation and damage processes in such emerging materials. Specifically, we are interested in the constitutive modeling of multi-phase materials, to include the interactions associated with the material microstructure, and the onset and evolution of damage as a time-dependent process. The unprecedented levels of reliability demanded of these future systems will also require a fundamental understanding of the response of structural materials to very high temperatures and severe temperature gradients and to high energy bombardment. Research issues include transient dynamic thermo-mechanical modeling, damage and failure development, life prediction and associated diagnostic techniques.

To focus attention to these issues we have successfully presented to the senior Air Force research management a new research initiative, entitled "Mesomechanics: The Microstructure-Mechanics Connection" (Ref 4, 5). The term "mesomechanics" is intended

to describe an area of research which bridges the microstructure-property relationship of materials with non-continuum mechanics. It expresses our belief that real progress in this endeavor can only come about by fostering a closer collaboration between a number of disciplines such as engineering mechanics, material science, applied mathematics, chemistry, etc. Quite contrary to the traditional approaches which seek to develop constitutive models from phenomenological observations of materials behavior, mesomechanics seeks to apply mechanics principles to the microstructural constituents of multi-phase materials, thus placing the microstructure-properties relationship on a quantitative basis.

The difficulties associated with the development of new approaches which can effectively deal with multiphase materials cannot be overstated. A number of significant obstacles must be overcome. Most prominent among them appear to be (i) the mathematical or geometrical description of the evolving material microstructure (which will serve as a common language among all disciplines involved), (ii) correct identification of local deformation and failure mechanisms and associated criteria for local equilibrium instability (which will guide the mechanics modeling of the physical processes), (iii) in-situ characterization of the mechanical properties of the constituents and interfaces/interphases, and (iv) large-capacity, very fast computational systems and algorithms.

Progress in all of these areas is needed before the aerospace systems projected for the early 21st Century can become a reality. The Air Force is promoting interdisciplinary cooperation at the basic research level as the most promising avenue to breakthroughs in understanding and predicting the behavior of evolving structural composites.

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1. Schriever, B., addressing the NATO Defense College, Paris, France, 15 June 1964.

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4. Haritos, G. K., et al, "Mesomechanics: the Microstructure-Mechanics Connection", Proceedings of 28th Structures, Structural Dynamics and Materials Conference, Part 1, AIAA, New York, 1987, pp 812-818.

5. Haritos, G. K., et al, "Mesomechanics: the Microstructure-Mechanics Connection", International J. Solids Structures, in press.

BRIEFER:

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Air Force Office of
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(202) 767-0463

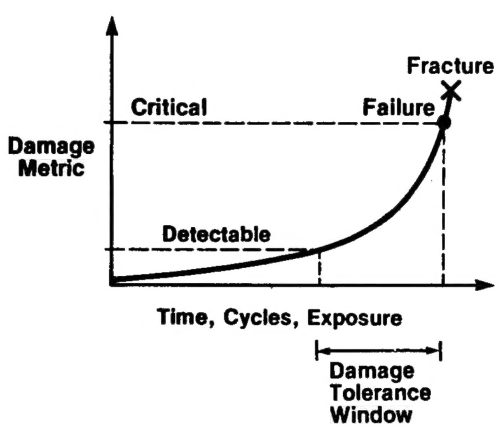
PROGRESS AND FUTURE CHALLENGES IN THE MECHANICS OF COMPOSITES

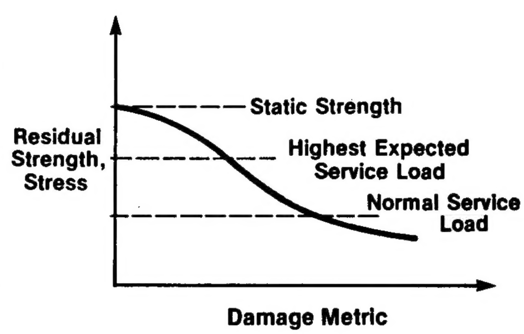
13th ANNUAL MECHANICS OF COMPOSITES REVIEW

BAL HARBOUR, FLA 2 NOV 88

Air Force Basic Research
Aerospace Sciences

Overview of Durability





Relevance:

- Damage Tolerance
  —Slow Growth
  —Fail Safe
- Retirement for Cause
- Durability
- Economic Life

Air Force Basic Research
Aerospace Sciences

2302/B2: STRUCTURAL DURABILITY

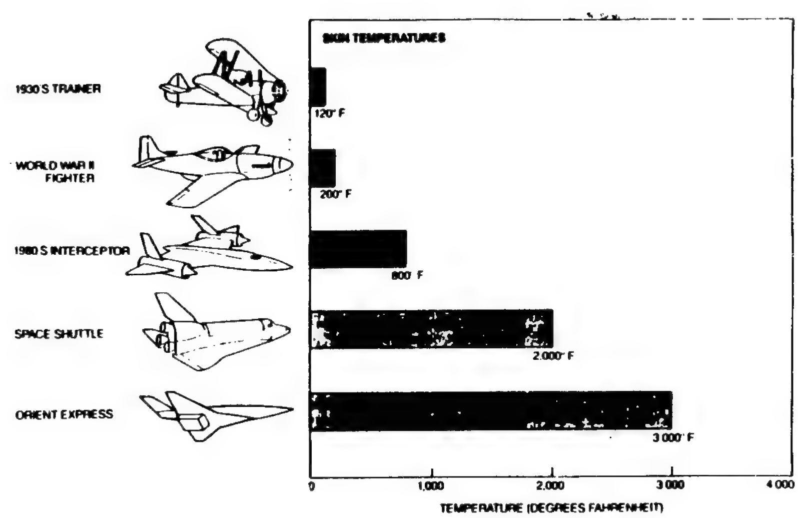
Current Research Philosophy

- **Classical**
  - Use continuum mechanics to represent material response
  - Failure theories based on the state of stress at a point

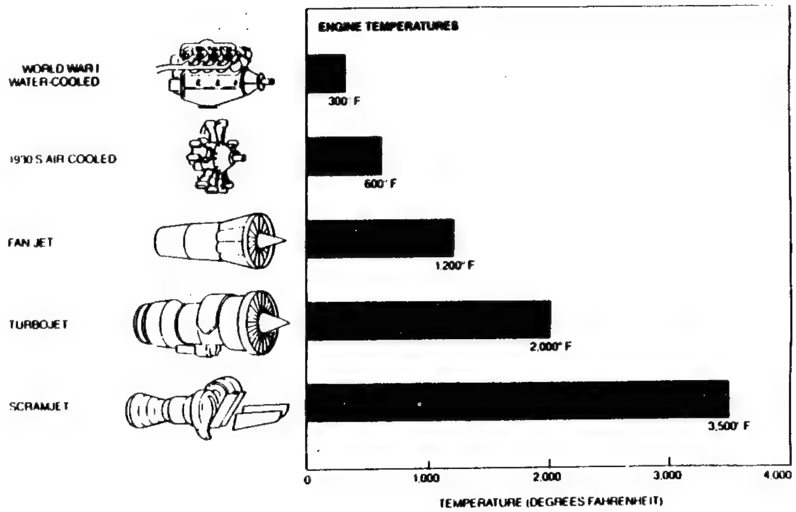
and Program Objectives

- **Develop Non-Continuum Mechanics for Multiphase Materials**
  - Use a heterogeneous representation to capture the effects of microstructural feature interactions
  - Describe development of damage as a load-, space-, and time-dependent process (non-local failure criteria)

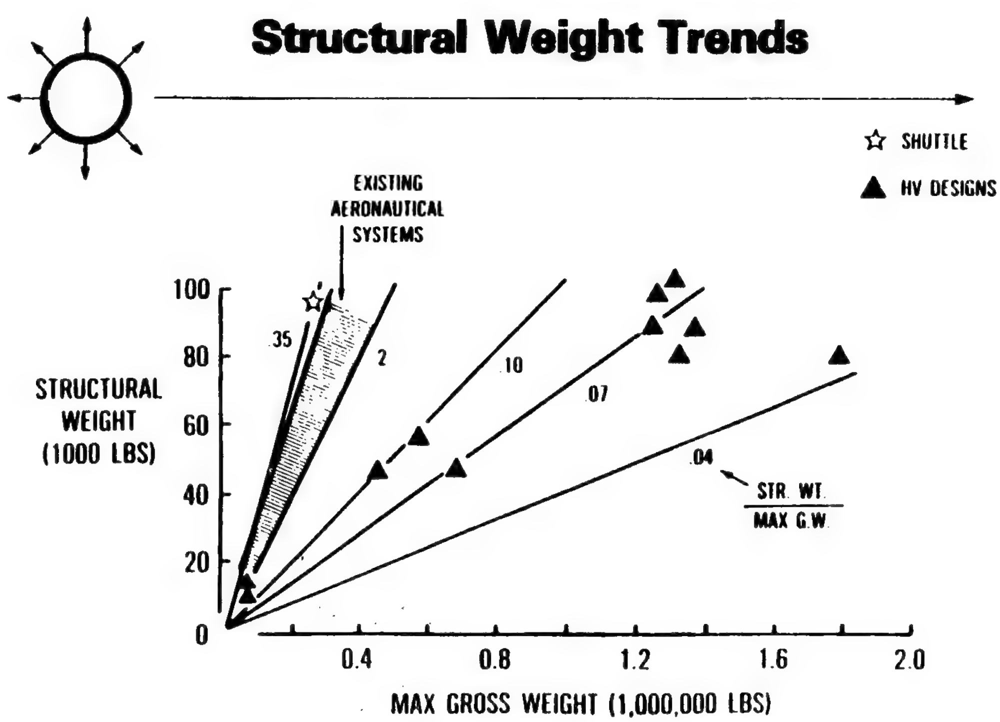
TRENDS IN AIRCRAFT SKIN TEMPERATURES



TRENDS IN ENGINE TEMPERATURES



Structural Weight Trends



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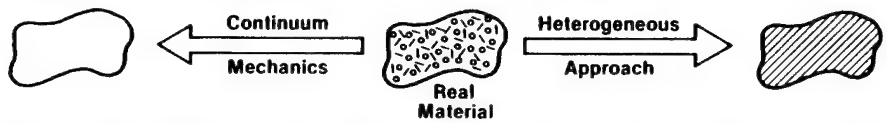
Mesomechanics: Constitutive Material Behavior

- ***CURRENT APPROACH***
- ***GOAL***

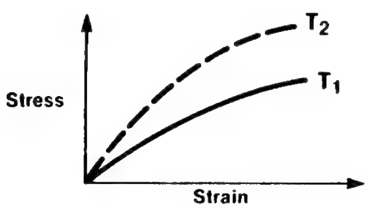


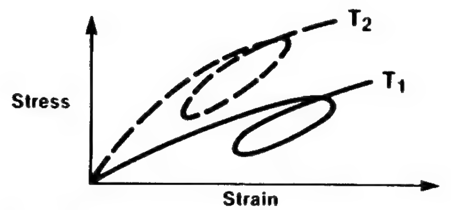
Current approach:

- **Failure Criteria Damage-Independent**
- **Unable to Predict Multiple-Failure Modes**
- **Cannot Predict Interactions Among Damage Micromechanisms**

Goal:

- **Damage Evolution Included in Constitutive Description**
- **Failure Criteria Based on Damage State**





Include Time and Damage in Material Behavior

PAYOFF: **Improved Durability, Improved Materials**

ROLE OF SOLID MECHANICS

Now	Future
"Bring Materials to Structures"	"Bring Structures to Materials"
material behavior + mechanics = structural theories	micro-structure + mechanics = material behavior

Material Behavior:

1. Deformation (constitutive equations)
2. Failure (strength theories)

CHALLENGES AND OPPORTUNITIES

CHALLENGES

- Ability to probe material at an ever decreasing scale
- Non-destructive, 3D observation of micromechanisms in real time
- Interactive analytical/experimental modeling via computer simulation
- Willingness to cross disciplinary boundaries

OPPORTUNITIES

- Consistent and coordinated advances in all fields related to materials and structures
- Material microstructures engineered (at will?) for desired performance

LIMITATION OF MICROMECHANISMS

- Geometrical/mathematical representation of microstructure
- Identification of local deformation and failure mechanisms
- In-situ characterization of constituents/interfaces
- Criteria for local equilibrium instability
- Large/fast computers and algorithms

Air Force Basic Research
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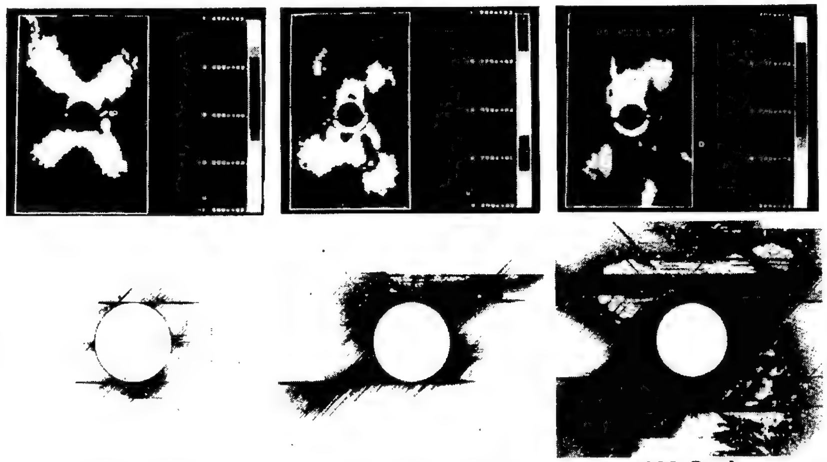
Differential Infrared Thermography Used to Determine Near-Tip Stress Distribution in Composites

Thermal Maps

Radiographs



~200 Cycles ~5,000 Cycles 10,000 Cycles

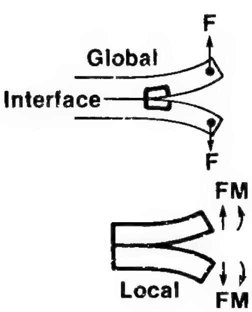
Research

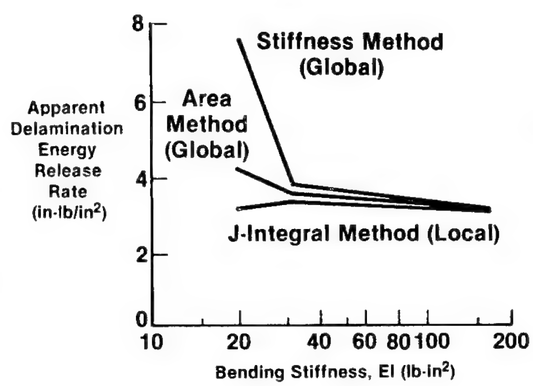
Analytical — Experimental Approach to Describe Damage Evolution

Ken Reifsnider/VPI

Air Force Basic Research
Aerospace Sciences

Global Data Analysis Gives Erroneously High Estimates of Delamination Energy in Composites





- Local (J-Integral) Method Captures Crack Energy Only
- Global Methods Include Bending Energy

Research

Difference Between Global and Local Analyses Can Be Used to Quantitatively Describe Remote Damage

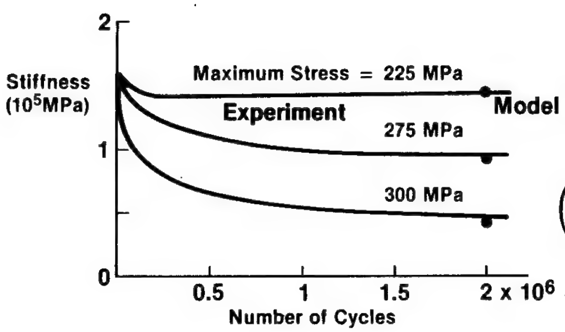
Dick Schapery/Tx A&M

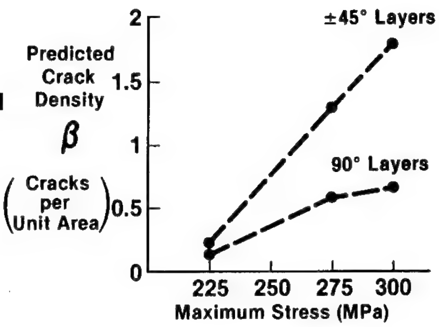
Air Force Basic Research

Aerospace Sciences

Shakedown-Damage Model Predicts Loss of Stiffness Due to Fatigue Damage for Metal-Matrix Composites

Boron — Aluminum $[0, \pm 45, 90, 0, \pm 45, 90_{1/2}]_s$





$$\text{Stiffness, } E = \frac{E_o}{1 + \beta \cdot g}$$

E_o — Initial Stiffness; β — Damage Parameter

$\beta = \beta$ (Stress Range, Stress Ratio, Cycles, Material Properties)

$g = g$ (Stress Range, Stress Ratio, Laminate)

- ***Research***

  Damage Evolution as a Constitutive Behavior of Composite

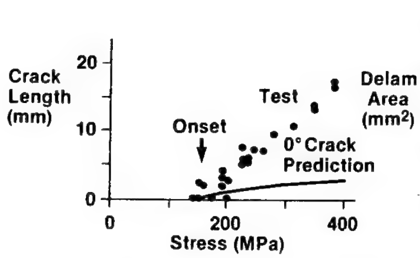
George Dvorak/RPI

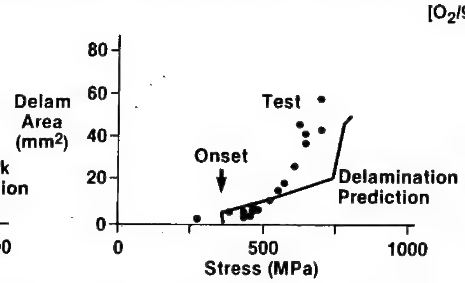
Air Force Basic Research

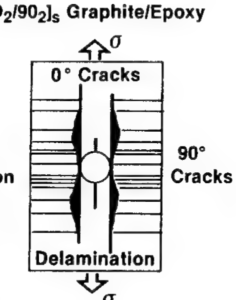
Aerospace Sciences

3D Finite Element Simulation Correctly Predicts Onset of Each Mode of Matrix Cracking in Composites, But Does Not Account for Interactions

$[0_2/90_2]_s$ Graphite/Epoxy







Simulation is Based on Theory of Ply Elasticity and on Concept of Linear Elastic Fracture Mechanics

- 0°-Layer Splitting Prediction Based on 0° Cracks Only
- Delamination Prediction Assumes All Matrix Cracking Completed

Research

Incorporate Cracking Mode Interactions in Simulation

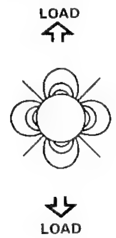
Wang/Drexel

Air Force Basic Research
Aerospace Sciences

New Micro-Mechanical Analysis Accurately Predicts Dynamic Stress Patterns in Composites

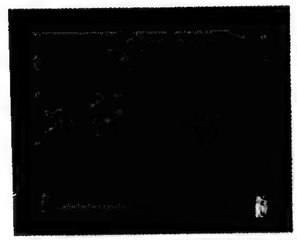
Example: Cyclic Loading of Notched $[45/90/-45/0]_S$ Graphite/Epoxy

Classical Mechanics Prediction

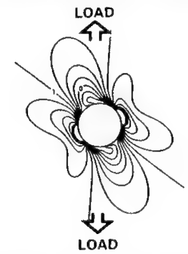


45° Fiber Orientation

Test (Thermography)



Micromechanics Prediction



$$\Delta T = -\frac{T_o}{C_\epsilon}\, \alpha_{kl}\, C_{ijkl}\, \epsilon_{ij}$$

Specific Heat (C_ϵ); Thermal Exp Coef (α_{kl}); Stiffness Tensor (C_{ijkl}); Strain Tensor (ϵ_{ij})

Perform Micro-Mechanical Analysis

- Apply Classical Equation to Each Constituent Material
- Use Micromechanics to Compute Strains In Each Material
- Use Average of ΔT in Fiber and Matrix Pointwise for Prediction

$$\Delta T = D_{13}\,\epsilon_1^I + D_{14}\,\epsilon_2^I$$

In-Plane Strains (ϵ_1^I, ϵ_2^I)

Elastic Material Constants (D_{13}, D_{14})

Relevance

Allows Accurate Computation of Residual Strength and Life

Reifsnider/VPI & SU

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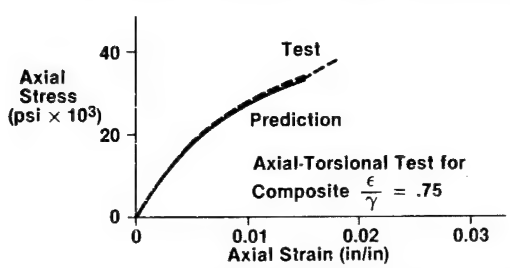
Potential Energy Method for Prediction of Constitutive Behavior Can Potentially be Extended to Account for Growing Damage

- Effect of Torsion on Axial Stress Accurately Predicted



$$\sigma = \frac{\partial}{\partial \epsilon}\int_0^\gamma \tau \, d\gamma + \sigma_0$$

Axial Stress (σ); Axial Strain (ϵ); Shear Stress (τ); Shear Strain (γ); Axial Stress at Zero Torsion (σ_0)

- Existence of Valid Work Potential Demonstrated

$$W_T = W_E + W_F$$

Total Work (W_T); Elastic Strain Energy (W_E); Fracture Work (W_F)

Necessary and Sufficient Condition for Existence of W_T

$$\frac{\partial \sigma}{\partial \gamma} = \frac{\partial \tau}{\partial \epsilon}$$

Research

Establish Forms of W_E, W_F in Terms of Basic Damage Parameters, e.g. Transverse Matrix Cracks. Delamination, etc.

Schapery/Texas A&M

Air Force Basic Research

Aerospace Sciences

FUTURE TRENDS IN TASK 2302/B2

Science Area	Trend	Decrease	Increase
Stochastic Methods	↗	Statistical Treatments	Evolutionary Methods (Stochastic, Probabilistic)
Crack Tip Fields	↗	Elastic Stress Intensity Approach	Damage Process Zone Observations/Models
Fatigue Crack Growth	↘	Predictions Based on Empirical/Statistical Models	Damage Mechanisms Identification and Interactions
Experimental Methods	↗	Room Temp,. Destructive, Macroscopic Measurements	High Temp, Non-Destructive, High Resolution, Sub-Micron Scale, Real Time
Failure Processes	↗	Linear Elastic Fracture Mechanics, Non-Interacting Failure Modes	Damage-Evolution Based Criteria
Damage Mechanisms	↗	Phenomenological, Continuum, Qualitative Description	Microstructural, Physically Founded, Quantitative

FAILURE MODES IN BRITTLE MATRIX COMPOSITES

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Materials Laboratory (AFWAL/MLBM)
Air Force Wright Aeronautical Laboratories
Wright-Patterson Air Force Base, Ohio 45433-6533

ABSTRACT

Fiber reinforced ceramic and glass-ceramic matrix composite materials are receiving a great deal of consideration for use in high temperature structural applications. These materials belong to a larger class which we call brittle matrix composites (BMC), characterized by matrices that are stiff relative to the fibers and exhibit low strain to failure. Furthermore, the fiber-matrix interface may be imperfect, either intentionally to provide subsequent resistance to initial matrix cracking or unintentionally caused by processing damage. Very high residual stresses may also be present in these materials.

In this work, we shall review the progress of our internal research program to study the mechanical behavior of BMC in integrated analytical-experimental effort. Analytical modeling work includes the development of an evolutionary model that consists of a concentric cylinder representative volume element in which fiber, matrix, and multiple coatings are recognized. The coatings may be introduced to modify the overall composite behavior, promote compatibility between constituents, or provide oxidation protection. For theoretical analysis, interphase (or interlayer) regions between fiber and matrix can be represented by coatings. The model permits the prescription of an arbitrary uniform state of stress at infinity and fibers oriented in N directions. Upper and lower bounds on composite moduli and effective thermal expansion and conductivity, as well as detailed stress fields within the constituents, are products of the model. Preliminary examination of the use of coatings to represent debonding or other constituent material damage is discussed.

Other analytical efforts are underway to provide more realistic representation of observed damage mechanisms in BMC, such as transverse matrix cracks that bridge the fibers, fiber breaks, and interfacial debonds.

Experimental work consists of careful detailed examination of damage progression in realistic BMC and also in model systems having more reproducible architecture and material properties. This investigation has produced significant evidence refuting conventional assumptions regarding both the nature of the initial damage and the corresponding stress level in the tensile loading of unidirectional composites parallel to the fiber direction. Observations of the transverse tensile response of BMC and comparison to analytical models reflect the drastic influence of interfacial weakness on composite stiffness and the potential significant impact on BMC laminate response.

Finally, some thoughts on the important issue of interface characterization will be presented.

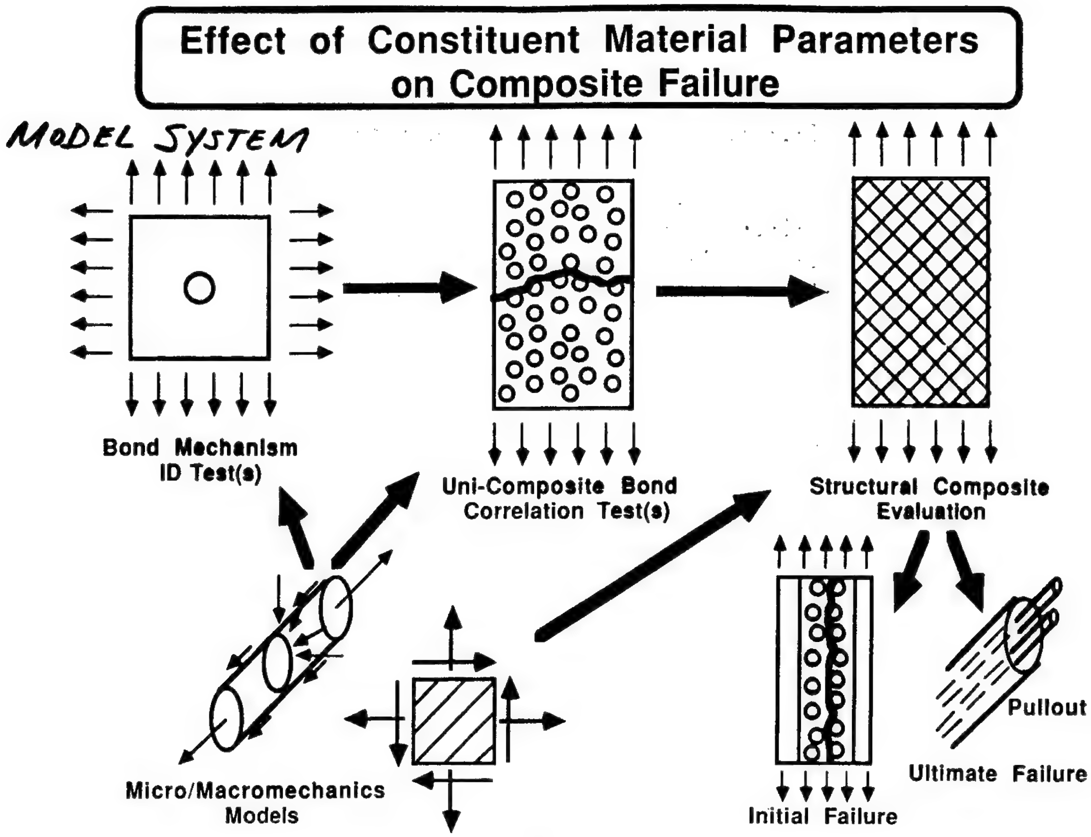
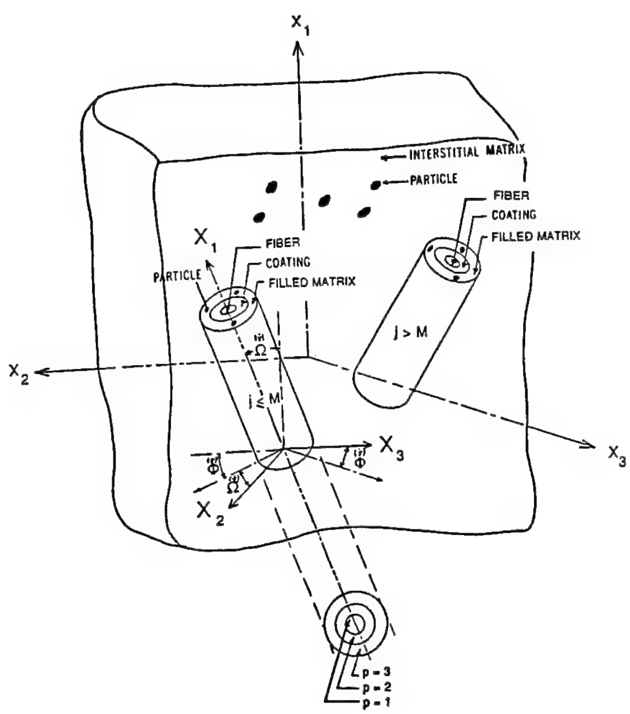


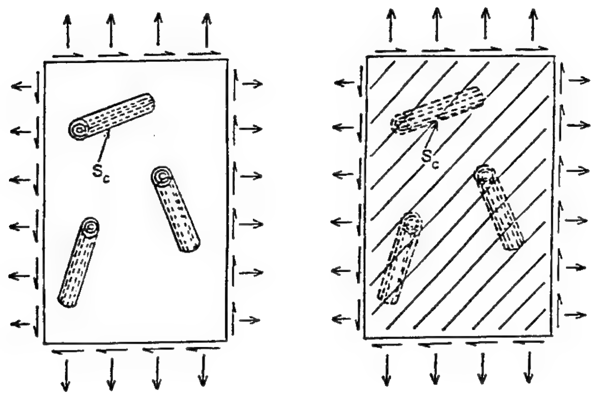
FIGURE 1

FIGURE 2 NOT REPRODUCIBLE



NDSANDS MODEL

FIGURE 3



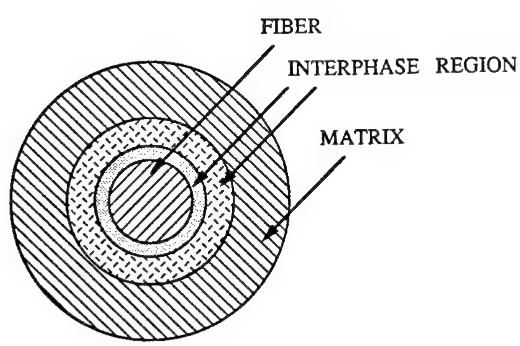
COMPOSITE EQUIVALENT HOMOGENEOUS BODY

$$\bar{\varepsilon}_{ij} = \overset{0}{\varepsilon}_{ij} \quad => \quad \overset{(j)}{u}_i = \overset{0}{\varepsilon}_{ik} \overset{(j)}{x}_k \text{ on } \overset{(j)}{S}_c$$

or

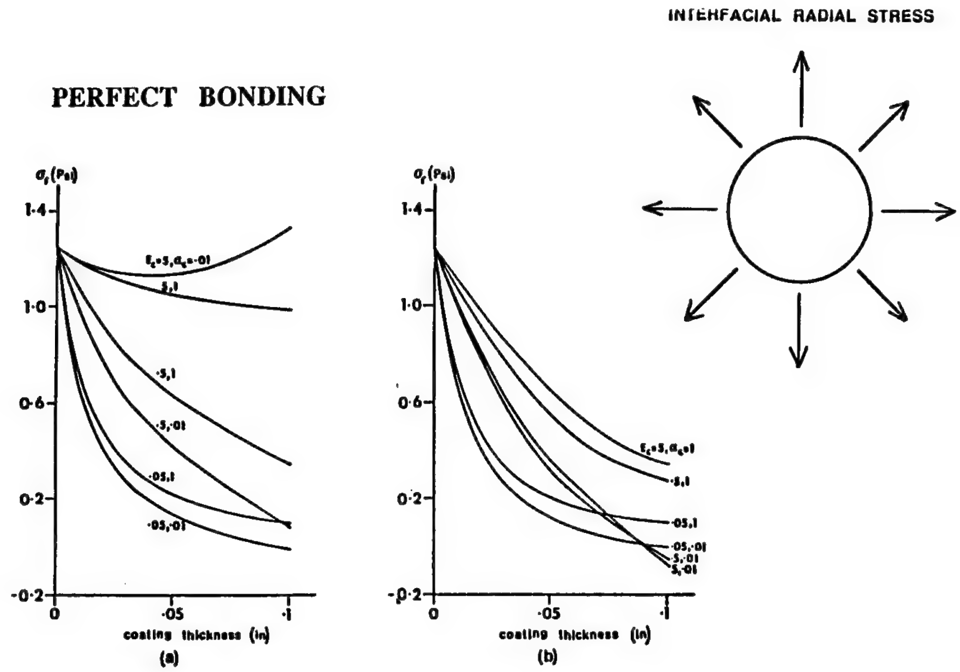
$$\bar{\sigma}_{ij} = \overset{0}{\sigma}_{ij} \quad => \quad \overset{(j)}{T}_i = \overset{0}{\sigma}_{ik} \overset{(j)}{n}_k \text{ on } \overset{(j)}{S}_c$$

FIGURE 4



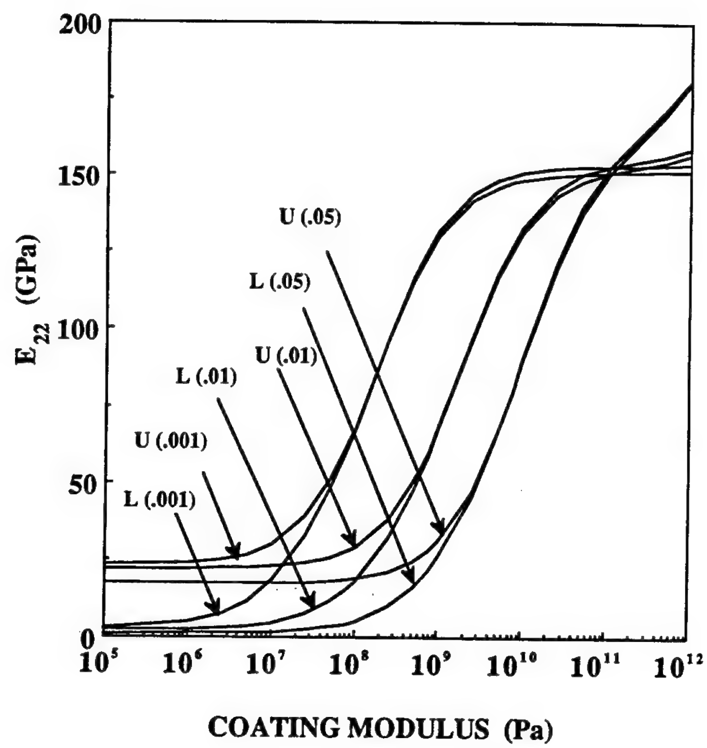
* ELASTIC STIFFNESS
* CONSTITUENT MATERIAL STRESSES
* INITIAL DAMAGE
* THERMAL EXPANSION AND CONDUCTIVITY

FIGURE 5



Radial stress, σ_r, at a) fiber-coating interface and; b) coating-matrix interface for Nicalon/BMAS composite under a uniform temperature change, $\Delta T = -1$.

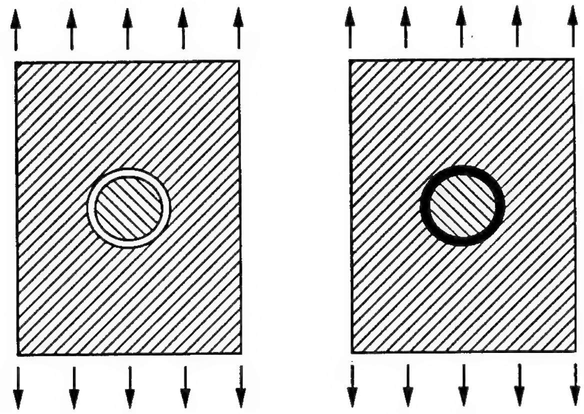
FIGURE 6



L : LOWER BOUND ; U : UPPER BOUND ;

(...) : RATIO OF COATING THICKNESS TO CYLINDER OUTER RADIUS

FIGURE 7



DEBONDING = SOFT INTERPHASE

(Thin coating layer of vanishing elastic modulus)

or SLIDING INTERFACE

($\tau_{r\theta} = \tau_{rz} = 0$; Continuity of u_r , σ_r)

FIGURE 8

Table 1: UPPER BOUND SOLUTION FOR EFFECTIVE ELASTIC MODULI FOR NICALON / BMAS COMPOSITE

Property	Case I	Case II	Case III	Case IV
E_{11}, Msi	23.572	23.572	6.084	6.149
E_{22}, Msi	22.166	13.387	13.310	2.769
G_{12}, Msi	8.791	0.891	0.891	0.891

CASE I : PERFECTLY BONDED INTERFACE

CASE II: SLIDING INTERFACE AND $(\varepsilon_z)_f = (\varepsilon_z)_m$

CASE III: SLIDING INTERFACE AND $(\varepsilon_z)_f = 0$

CASE IV: CYLINDRICAL VOID IN MATRIX

FIGURE 9

TRANSVERSE MODULUS AND STRENGTH

Matrix	Modulus, Msi Tension	Modulus, Msi Compr.	Tensile Strength psi
CAS	15.9	15.9	3,045
MAS/LZN	1.88	2.25	626
MAS/LZN	5.19	5.19	530
MAS/LZN	3.96	3.52	310
MAS/LZN	2.36	2.44	164
MAS/LZN	1.70	1.97	523
1723	10.25	10.25	226
1723	5.86	17.80	418
1723	1.11	1.18	447

FIGURE 10

NDSANDS RESULTS

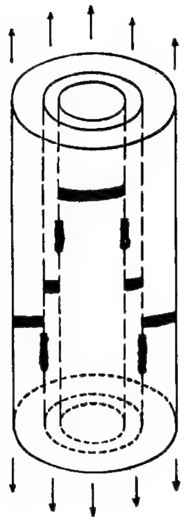
MAS/LZN with 40% VOIDS

E_U = 5.69 Msi

E_L = 2.08 Msi

FIGURE 11

BMC MODEL



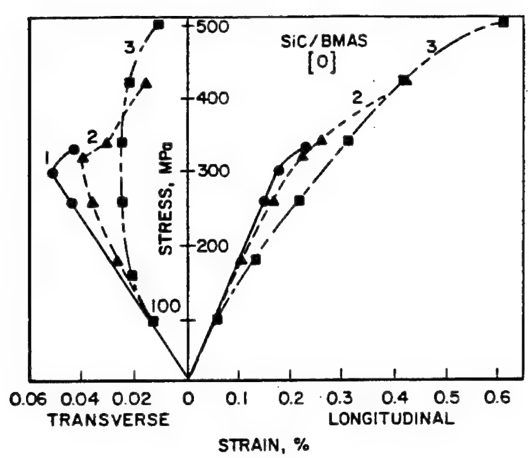
AXISYMMETRIC PROBLEMS

LOADINGS	FEATURES
AXIAL	6 NON-ZERO σ_{ij}
TORSION	MULTI-COMPONENT
HYDROSTATIC	DISC. INTERFACES
LONG. SHEAR	ANISOTROPIC MEDIA

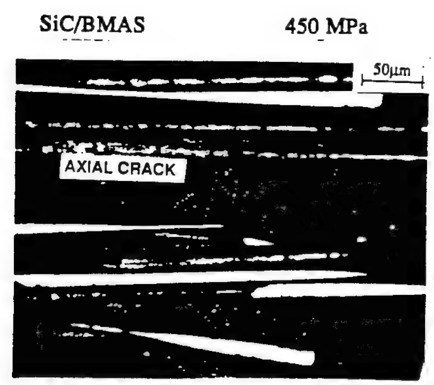
DAMAGE MODES

TRANSVERSE CRACKS
BONDED/DEBONDED/NT.
FRICTION INTERFACES

FIGURE 12



Stress-Strain Curves for SiC/BMAS.



Photomicrograph Showing Axial Cracks on a Tensile Specimen of SiC/BMAS.

FIGURE 13

SiC/LAS III

SEM Pictures Showing Interface Debonding.

FIGURE 14

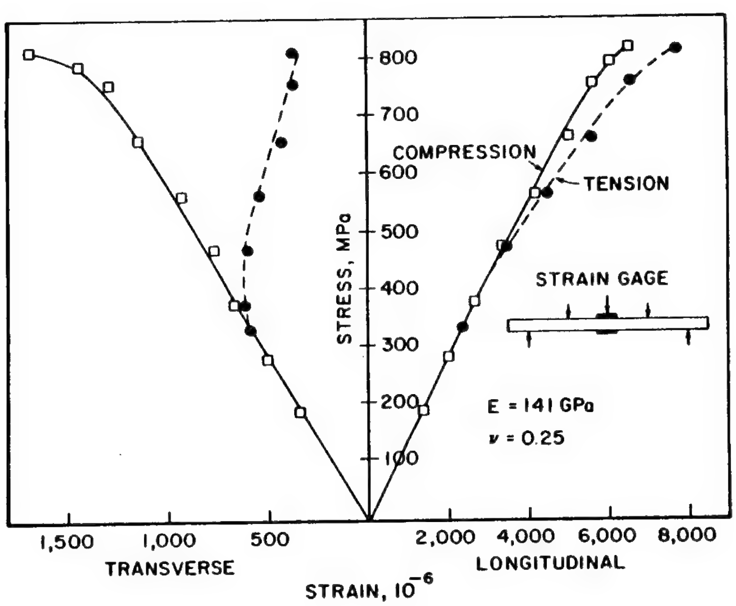
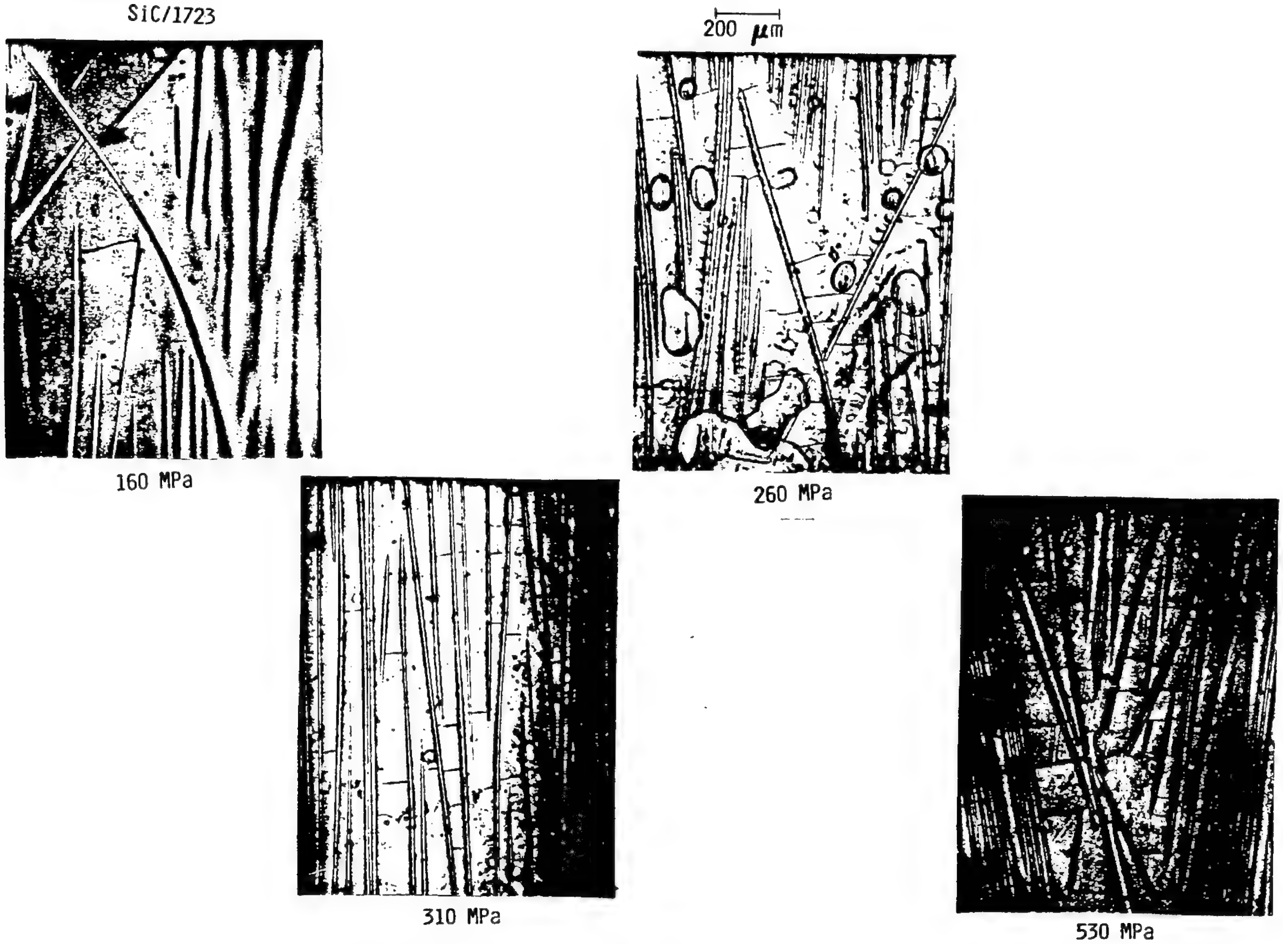
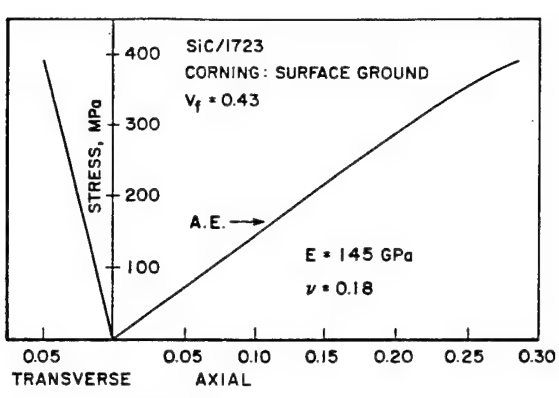


FIGURE 15

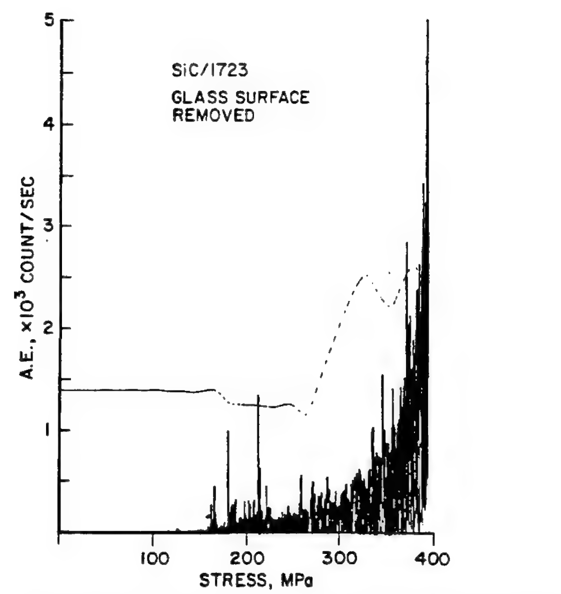


Photomicrographs Showing Matrix Cracks at Four Stress Levels.

FIGURE 16



Stress-Strain Curves for SiC/1723 Type B.



Acoustic Event Versus Stress for SiC/1723 Type B.

FIGURE 17

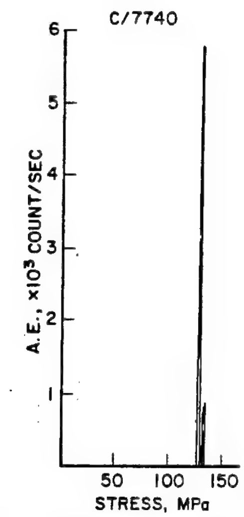


Figure 2. Acoustic Event Versus Stress for C/7740.

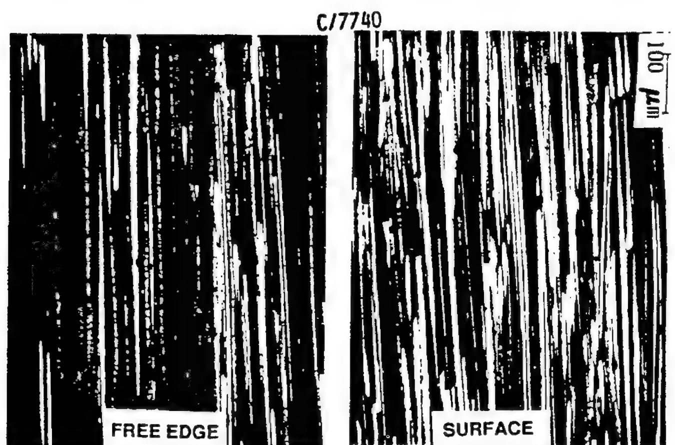


Figure 3. Photomicrographs Showing Matrix Cracks at the Stress 133 MPa.

FIGURE 18

	C/7740	SiC/1723 A	SiC/1723 B	SiC/CAS	SiC/BMAS	SiC/LASIII
FIBER VOLUME, %	42	65	43	40	39	43
PROPORTIONAL LIMIT, MPa	350	350	286	212	254	306
MATRIX CRACKING STRESS, MPa	123	201	160	132	-	195
(STRAIN, %)	(0.07)	(0.13)	(0.11)	(0.10)	-	(0.15)

FIGURE 19

MODEL SYSTEM APPROACH

A) CHOICE OF FIBER / MATRIX COMBINATIONS

* CHEMICAL COMPATIBILITY
* STABLE CHEMISTRIES
* HIGH STRENGTH
* TRANSPARENCY OF THE MATRIX
* VARIABLE RESIDUAL STRESS STATES

B) DEVELOPMENT OF PROCESSING TECHNIQUES

C) MEASUREMENT OF CONSTITUENT AND COMPOSITE PROPERTIES

D) NOVEL TEST METHODS TO STUDY FAILURE MODES

* STRAINING STAGE ON A OPTICAL MICROSCOPE
* USAGE OF SURFACE REPLICAS DURING TENSILE TESTING

FIGURE 20

COMPOSITE SYSTEMS

* SILICON CARBIDE MONOFILAMENT
  AVCO SCS-6 FIBER

* BOROSILICATE GLASS MATRICES
  CGW 7052, 9741, 7740, 7761

* HIGHER AND LOWER CTE COMPARED
  TO SCS-6 MONOFILAMENT

FIGURE 21

CASE I: $\alpha_f > \alpha_m$

CGW 7761 GLASS WITH AVCO SCS-6 SiC FIBER

* σ_r IS TENSILE AT FIBER-MATRIX INTERFACE
* DEBONDING AT FIBER-MATRIX AND AT CORE-FIBER INTERFACES
* RANDOM FIBER AND MATRIX CRACKING AT LOW STRESSES
* EXTENSIVE MATRIX SHATTERING AT HIGH STRESSES

FIGURE 22

Random fiber and matrix cracks in 7761/SCS - 6 fiber composite (100x)

Extensive damage (debonding) at fiber-matrix interface in 7761/ SCS - 6 fiber composite (200x)

FIGURE 23

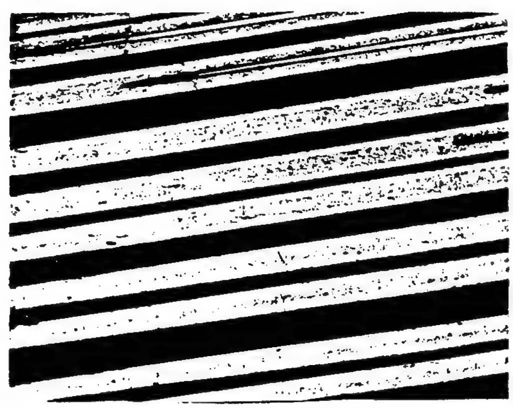
CASE II : $\alpha_f < \alpha_m$

CGW 7052 GLASS WITH AVCO SCS-6 SiC FIBER

* σ_r IS COMPRESSIVE AT FIBER-MATRIX INTERFACE
* NO DEBONDING AT INTERFACES
* MULTIPLE UNIFORMLY SPACED TRANSVERSE MATRIX CRACKING AFTER PROCESSING
* FIBER CRACKS ON LOADING CLOSE TO THE EXISTING MATRIX CRACKS

FIGURE 24

Multiple matrix cracking in 7052/ SCS - 6 fiber composite after processing (100x)



Fiber cracks under stress close to existing matrix cracks in 7052/ SCS - 6 fiber composite (100x)

FIGURE 25

ON THE ROLE OF DEFECTS IN FRACTURE OF COMPOSITES

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ABSTRACT

Recognizing a leading role of damage in fracture on various scales, we distinguish two extreme cases:

Cooperative Fracture (see Fig. 1). The intensity of damage formed as a response to the stress concentration at the tip of a propagating crack is much greater than the intensity of the pre-existing damage. Crack propagation is then inseparable from the evolution of the damage accompanying the crack. Although the microdefect population in the vicinity of the crack tip is random, the overall behavior is highly reproducible. Integral damage characteristics and the thermodynamics of irreversible processes are employed in the Crack Layer Theory [1] to model the phenomenon.

Solo Crack Fracture (see Fig. 2). A crack propagates through a pre-existing field of microdefects causing negligible changes to the field. The fluctuation of the microdefect field leads to the scatter of experimentally observed fracture parameters, such as critical crack length, critical load, etc. Crack Diffusion Model has been recently advanced to describe the phenomenon in probabilistic terms [2,3].

The Crack Layer (CL) Theory is based on the hypothesis of self-similarity of damage evolution. It reduces the evolution of the CL to simple translation, rotation and deformation of the damage zone. The rates of these elementary motions are treated as thermodynamic fluxes. The conjugate forces are introduced through the entropy production within the general framework of thermodynamics of irreversible processes. The relationship between the forces and the fluxes and conditions of CL instability which determine toughness are proposed. Comparison of the proposed model with experimental data on short glass fiber reinforced epoxy is presented.

Crack Diffusion Model (CDM) is developed to describe ideally brittle fracture (no accompanying damage or plasticity) which is controlled by a pre-existing field of microdefects. In relation to fracture, the defect population is characterized by a random field of specific fracture energy γ.

A set Ω of possible crack trajectories, reflecting statistical nature of brittle fracture, is introduced in CDM (see Fig. 3). Macroscopical fracture parameters are expressed through averaging over Ω.

The output of the CDM is twofold: a) it proposes a methodology of the statistical characterization of the γ-field (material property!) based on experimentally observed quantities such as crack arrest location, critical crack tip position and critical load distributions, b) it offers a methodology for prediction of the probability of failure for engineering structures.

Comparison of CDM with experimental data on short Kevlar fiber reinforced polyester is presented.

REFERENCES

1. A. Chudnovsky, J. Botsis and A. Moet, "Fatigue Crack Layer Propagation in Polystyrene," I and II, Intern. J. Fracture, 33, 1987, p. 263 and p. 277.

2. A. Chudnovsky and B. Kunin, "A Probabilistic Model of Brittle Crack Formation," J. Appl. Phys. 62 (10), 1987, pp. 4124-29.

3. A. Chudnovsky and B. Kunin, "A Probabilistic Model of Brittle Crack Formation, II," in preparation.

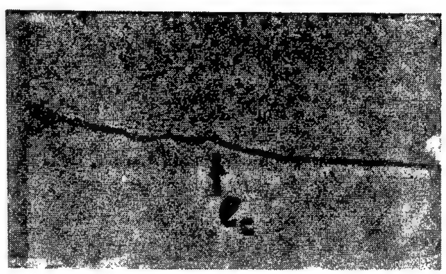
Extreme Cases in Fracture

Cooperative:

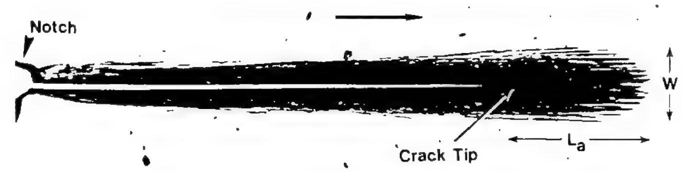
Fig. 1:

Solo crack:

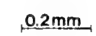
Fig. 2:



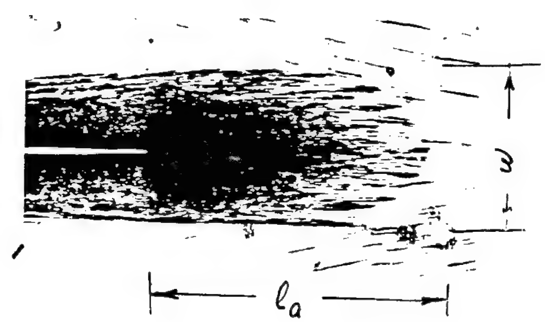
Crack Layer Model



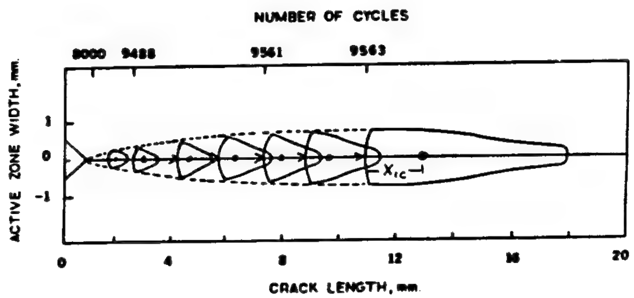
Damage density ρ : $[\rho] = \frac{m^2}{m^3}$; $\frac{kg}{m^3}$



ACTIVE ZONE V_A : $\dot{\rho}(x) > 0$



Kinematics of Crack Layer



$\dot{\ell}$ — crack speed

$\dot{\underset{\sim}{x}}_c$ — AZ "translation"

$\dot{\omega}$ — AZ "rotation"

$\dot{\underset{\sim}{V}}$ — AZ "deformation"

Fast and Slow Processes

$$\dot{\underset{\sim}{x}}_c = \dot{\ell}\, \partial_\ell \underset{\sim}{x}_c + \partial_t \underset{\sim}{x}_c$$

$$\dot{\omega} = \dot{\ell}\, \partial_\ell \omega + \partial_t \omega$$

$$\dot{\underset{\sim}{V}} = \dot{\ell}\, \partial_\ell \underset{\sim}{V} + \partial_t \underset{\sim}{V}$$

What are the Driving Forces ?

CL Driving Forces

$$\dot{S}_i^{global} = \dot{\ell} \cdot X_1^c + \frac{1}{T} \underbrace{\dot{D}}_{\partial_t x_1^c \cdot X_1^{tr} + \partial_t e \cdot X^{exp} + \partial_t d_{ij} X_{ij}^{dis}} \geq 0$$

Forces :

$$X = \frac{1}{T}\left(A - \gamma R\right)$$

Active Parts :

$$A_1^c = -\frac{\partial \Pi}{\partial e} \; ; \; A_1^{tr} = -\frac{\partial \Pi}{\partial x_1^c} \; ; \; A^{exp} = -\frac{\partial \Pi}{\partial e}$$

Resistive Parts :

$$R_1^{tr} = \int_{\Gamma^{tr}} \rho \cdot n_1 \, d\Gamma \; ; \; R^{exp} = \int_{V_A} \rho \, dV \; ; \; R_{ij} = \int_{V_A} \xi_{(i} \partial_{j)} \rho \, dV - R^{exp} \delta_{ij}$$

$$A_1^c = J_1^c$$

$$A_1^{tr} = J_1^A$$

$$A^{exp} = M$$

Energy Release Rates

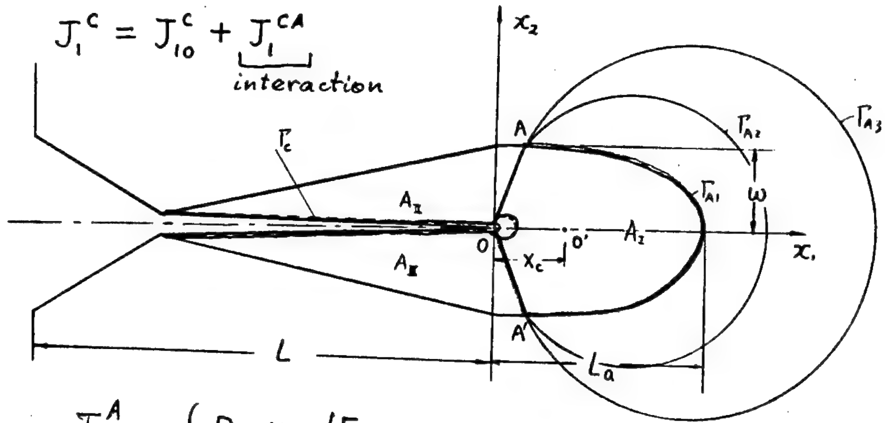
$$J_1 = \int_\Gamma P_{i1} n_i \, d\Gamma$$

$$M = \int_\Gamma \xi_j P_{ij} n_i \, d\Gamma$$

Energy Release Rates Evaluation

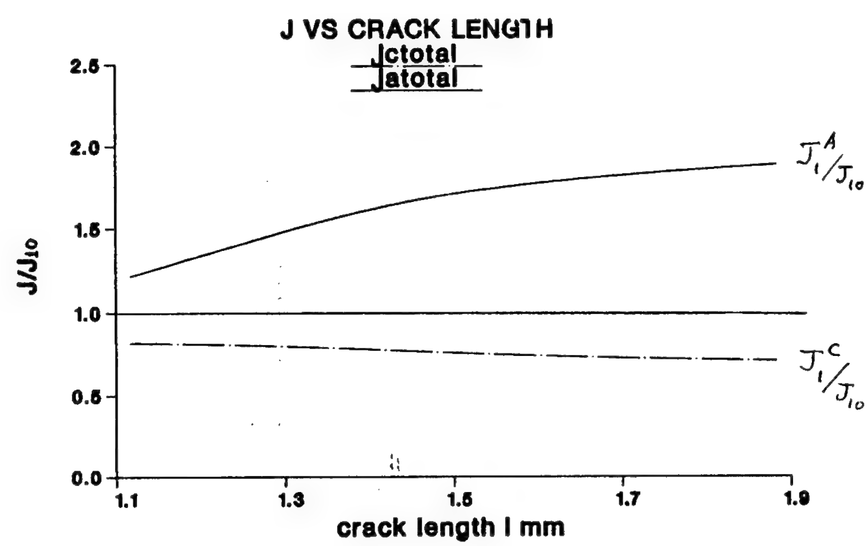
$$J_1^C = \int_{\Gamma_C} P_{i1}\, n_i \, d\Gamma$$

$$J_1^C = J_{10}^C + \underbrace{J_1^{CA}}_{\text{interaction}}$$



$$J_1^A = \int_{\Gamma_{A1}} P_{i1}\, n_i \, d\Gamma$$

$$J_1^A = J_{10}^A + \underbrace{J_1^{AC}}_{\text{interaction}}$$



Slow Crack Growth

Thermodynamic Stability :

$$\dot{\ell}\,\frac{dX^c}{dt} < 0 \;\Rightarrow\; \frac{d}{d\ell}X^c < 0 \;\Rightarrow\; \left(\frac{d}{d\ell}J_1^c < 0\right)$$

The Second Low :

$$\dot{S}_i^{glob.} = \underbrace{\dot{\ell}\cdot X^c}_{<0} + \frac{1}{T}\dot{D} \geq 0$$

a) $$\boxed{\dot{\ell} = \frac{\dot{D}}{\gamma R_1^c - A_1^c}}$$

Toughness :

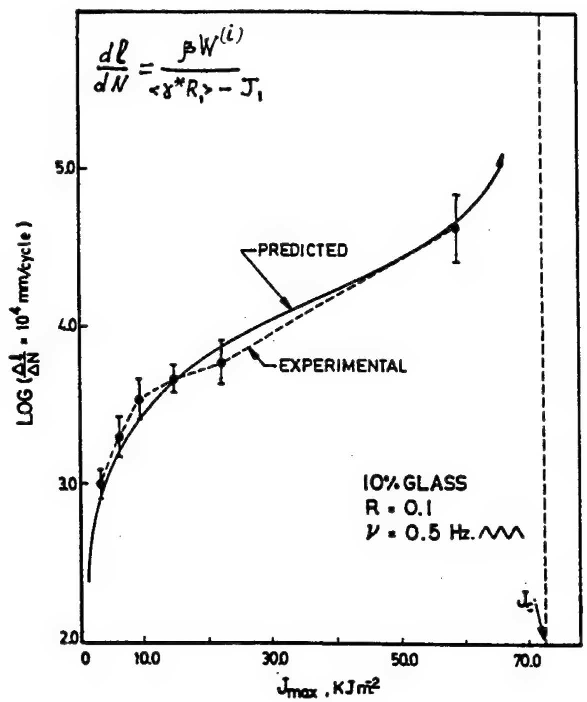
$$A_1^{c*} = \gamma \cdot R_1^{c*}$$

b) Governing Equations for Slow Process

$$\partial_t e \sim X^{exp}$$

$$\partial_t d_{ij} \sim X_{ij}^{dis}$$

Fatique Crack Growth



Toughness Characterization of Composites

$$J_{Ic} = \sum_{(K)} \gamma^{(K)} R_{(K)} = \underline{\left(\gamma \cdot R_1\right)}^{eff}$$

Practical Implications

(i) Characterization

$$J_{Ic} = \left(\gamma \cdot R_1\right)^{eff}$$

$$\Downarrow$$

$$R_1 = R_1(\underset{\sim}{\sigma}, \dot{\underset{\sim}{\sigma}}, T, \ldots)$$

creep – fatigue – impact

(ii) **Design of composite**

$$\gamma^K, R_K \Longrightarrow \left(\gamma R_1\right)^{eff}$$

Toughness Enhancement

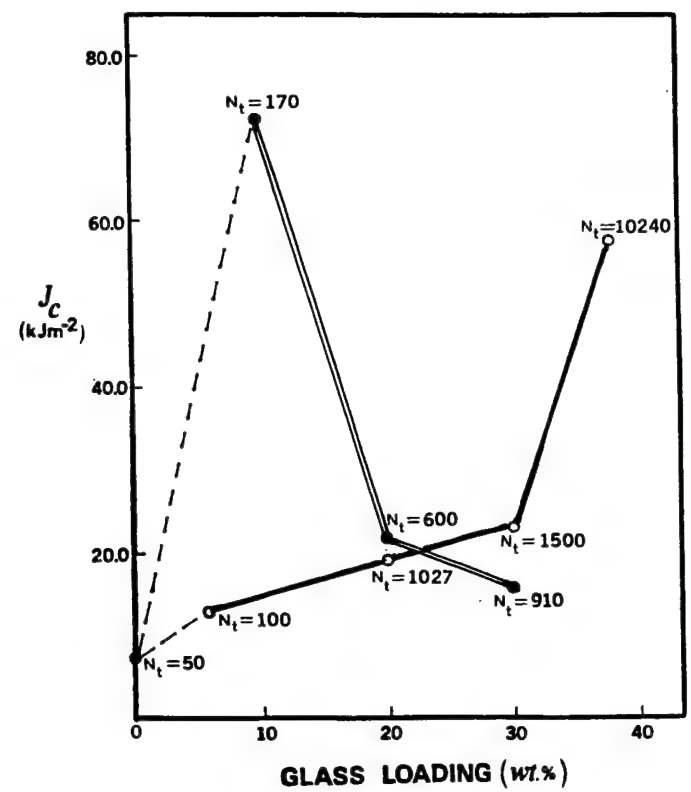
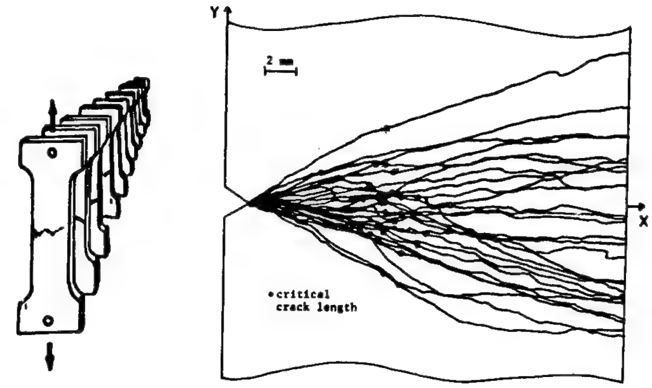


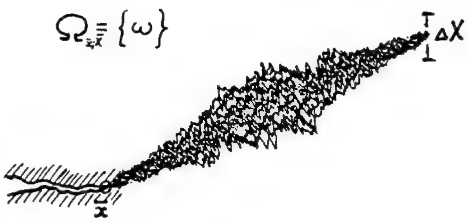
Fig. 3:

Ensemble of Possible Crack Trajectories



Crack Propagator

$\Omega_{\underline{x},\underline{X}} \equiv \{\omega\}$

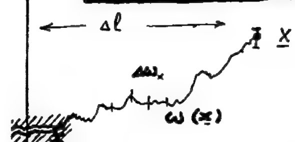


$$\mathbf{P}(\underline{x},\underline{X}) = \sum_{\Omega_{\underline{x},\underline{X}}} \mathbf{P}(\underline{x},\underline{X}/\omega)\cdot p\{\omega\}$$

$$\Downarrow$$

$$\int_{\Omega_{\underline{x},\underline{X}}} \mathrm{P}(\underline{x},\underline{X}/\omega)\, d\mu(\omega)$$

Conditional Propagator



$\tau_0 < \Delta\omega \ll \Delta l$

$$P(\underline{x},\underline{X}/\omega) = \lim_{N\to\infty} P\Big\{\bigcap_{k=1}^{N}\big(J_1^{(k)}\Delta\omega_k > 2\gamma\,\Delta\omega_k\big)\Big\}$$

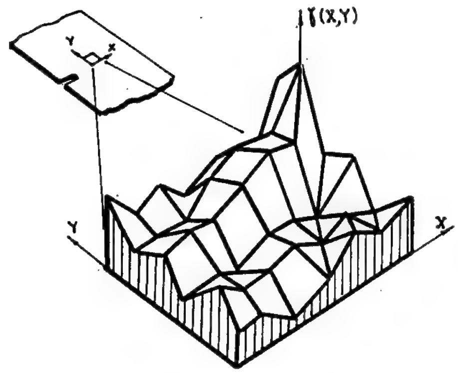
Griffith' Condition

$$\Downarrow$$

$$P(\underline{x},\underline{X}/\omega) = \exp - \int_{x_1}^{X_1} P\{2\gamma(\xi) \geq J_\omega(\xi)\}\,\frac{d\xi}{\tau_0}$$

probability of crack arrest within $\Delta\xi$

RANDOM FIELD OF STRENGTH $\gamma(x,y)$.



Point-wise γ-distribution along ω:

$$F(\gamma) = 1 - e^{-\left(\frac{\gamma - \gamma_{min}}{\langle\gamma\rangle - \gamma_{min}}\right)^{\alpha}}$$

r_0 – correlation distance

Crack Propagator

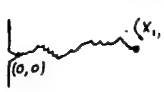
$$P\{c.a.\ \text{in}\ d\xi\} = V(\xi,\omega)\,\frac{d\xi}{r_0}$$

$$V(\xi,\omega) = \exp - \left\{\left[\frac{J_1(\xi,\omega) - 2\gamma_{min}}{2(\langle\gamma\rangle - \gamma_{min})}\right]^{\alpha}\right\}$$

$$P(\underline{x},\underline{X}) = \int_{\Omega_{\underline{x},\underline{X}}} \left[\exp - \int_{x_1}^{X_1} V(\xi,\omega)\,\frac{d\xi}{r_0}\right] d\mu(\omega)$$

Applications

- Crack Arrest Location

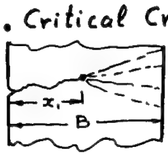


$\underline{C.P.}$ $\quad P(X_1,X_2) = P(0,0;X_1,X_2)$

$$\underline{P_a(X_1,X_2) = P(X_1,X_2)\cdot V(X_1,X_2)\cdot\frac{1}{r_0}}$$

$$P(X_1,X_2) = \int_{\Omega_{0,X}} \left[\exp - \int_0^{X_1} V(\xi,\omega)\,\frac{d\xi}{r_0}\right] d\mu(\omega)$$

- Critical Crack Tip Location



$$P(x_1,x_2) = \int_{-\infty}^{\infty} P(x_1,x_2;B,X_2)\,dX_2$$

$$P(x_1,x_2;B,X_2) = \int_{\Omega_{\underline{x},B,X_2}} \left[\exp - \int_{x_1}^{B} V(\xi,\omega)\,\frac{d\xi}{r_0}\right] d\mu(\omega)$$

$$\underline{P_c(x_1,x_2) = P_a(x_1,x_2)\cdot P(x_1,x_2)}$$

Evaluation of $P(x_1,x_2)$, $P(X_1,X_2)$
(Diffusion' Approximation)

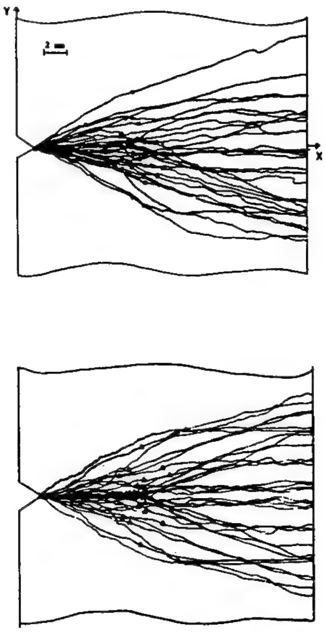
Stable configuration

$$\begin{cases} \dfrac{\partial P(X_1,X_2)}{\partial X_1} = \dfrac{D}{2}\,\dfrac{\partial^2 P(X_1,X_2)}{\partial X_2^2} - V(X_1,X_2)\cdot P(X_1,X_2) \\ P(0,X_2) = \delta(X_2) \\ P(X_1,X_2) \to 0 \quad \text{as } X_2 \to \pm\infty \quad (X_1 \geq 0) \end{cases}$$

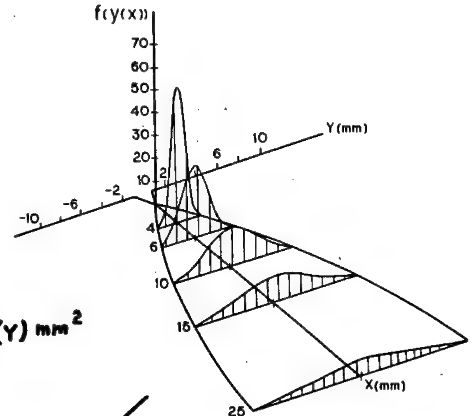
Unstable configuration

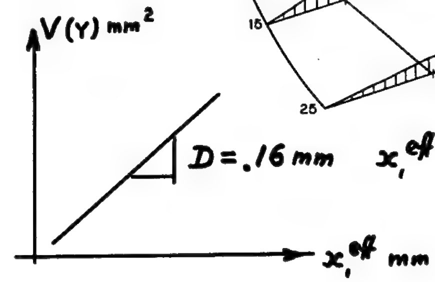
$$\begin{cases} \dfrac{\partial P(x_1,x_2)}{\partial x_1} = -\dfrac{D}{2}\,\dfrac{\partial^2 P(x_1,x_2)}{\partial x_2} + V(x_1,x_2)P(x_1,x_2) \\ P(B,x_2) \equiv 1 \\ \dfrac{\partial P(x_1,x_2)}{\partial x_2} \to 0 \quad \text{as } x_2 \to \pm\infty;\ (X_1 \leq B) \end{cases}$$

Crack Diffusion Analysis



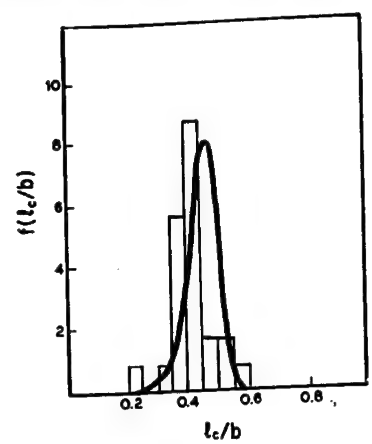
Crack Diffusion Coefficient





$$x_1^{eff} = \frac{E \cdot J_1}{\pi \cdot \sigma^2}$$

Distribution of critical crack lengths

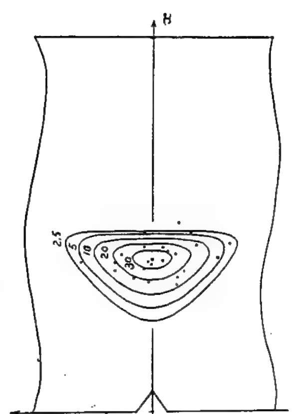


$\gamma_o = 3.7 \quad kJ/m^2$

$\alpha = 2.9$

$$F(\gamma) = 1 - exp[-(\frac{\gamma}{\gamma_o})^\alpha]$$

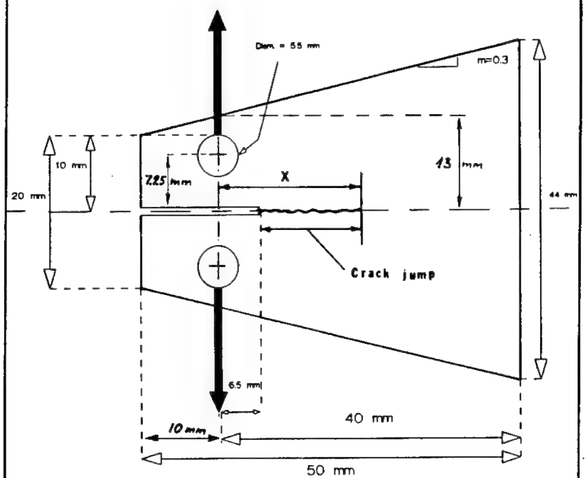
Probability Density of Critical Crack Tip Location

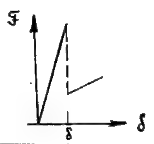


$\chi^2 = 2.7 \qquad Q(\chi^2|\nu) = .75 \qquad$ (comp. to .1)

Crack Arrest Experiment

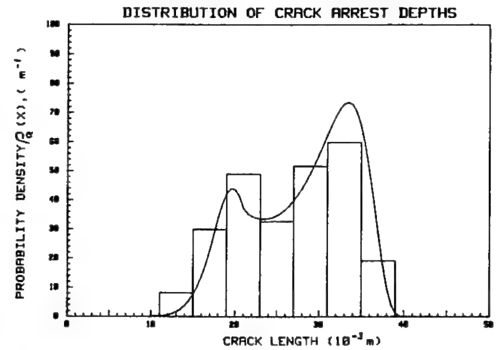
Tapered Specimen Geometry





(δ, X)

$$P_a(x) = \frac{1}{r_0} \exp\left[-\left(\frac{J_\omega(x)}{2\gamma_0}\right)^{\alpha}\right] \times \int_{\Omega_{0,x_1}} \left[\exp - \int_0^{x_1} \exp - \left(\frac{J_1(x,\omega)}{2\gamma_0}\right)^{\alpha} \frac{dx}{r_0}\right] d\mu(\omega)$$



$2\gamma = 360\ J/m^2$

$\alpha = 0.6$

Fracture controled by preexisting damage "conforms to the laws of probability, but the probability itself is propagated in accordance with the Law of Causality"

3D STRESSES AT THE EDGE OF A CIRCULAR HOLE IN A COMPOSITE LAMINATE

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ABSTRACT

It has long been recognized that delamination is one of the most important failure modes in laminated composite structures. The growth of a delamination may result in a reduction of strength and stiffness of the laminate. The identification, therefore, of such locations in a composite structure is of great interest to the designer. For example, the stresses at the intersection between a free edge and an interface may be singular. Recent investigations [1,2] show that a stress singularity exists for certain types of laminates. The condition is even worse if the free surface represents a cylindrical cut out. Knowledge of the stress field in such areas is of great importance to the designer. Unfortunately, the problem is 3D in nature and is very difficult to solve. Moreover, the presence of a material interface adds to the complexity of the problem.

The author in this paper investigates the 3D stress field in the neighborhood of a bonded interface and the free edge of a hole in a laminated composite plate. The laminates are assumed to be transversly isotropic with a $0^{\circ}/90^{\circ}/0^{\circ}$.... stacking sequence. Perfect bonding is assumed at the interface. Far away from the hole, the plate is subjected to a uniform extensional load (see fig. 1).

A 3D analytical solution applicable to the region where the interface meets the free-of-stress surface of the hole is constructed. The analysis shows the stress field to be singular for certain material constants and of the form:

$$\sigma_{ij}^{(m)} = \rho^{\alpha} f_{ij}(\theta,\phi) \qquad ; \qquad m = 1,2 .$$

Moreover, the singularity exponent α is shown to be a function of the position angle θ and of the material constants. A result that is compatible with experimental observations. The special case where the laminates are of homogeneous and isotropic material is also examined. For this case, the exponent α is independent of θ and its value is precisely the same as that of the corresponding two quarter spaces bonded along the positive x-axis [3].

Finally, the 3D stress field of an inclusion embedded in a plate of an arbitrary thickness 2h is also discussed for soft as well as rigid inclusions.

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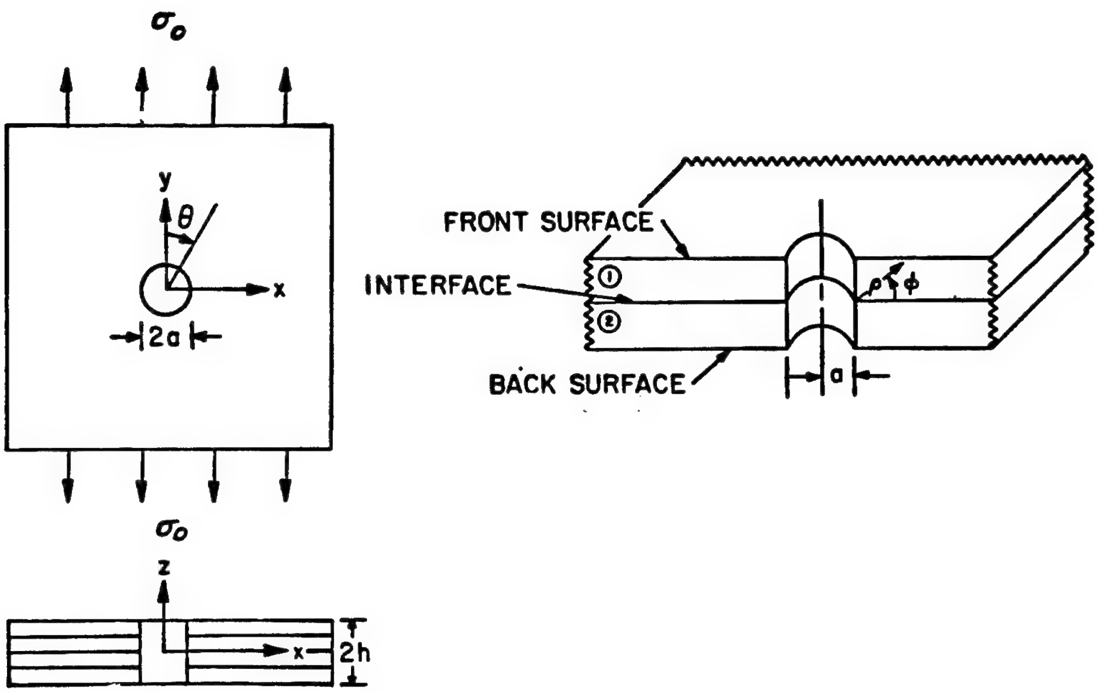
LAMINATED COMPOSITE PLATES--INTERLAMINAR STRESSES

Model: 3D linear elasticity
Material: Transversely Isotropic Laminae
Method of Solution: Analytical
Objective: Local Stress Fields at Position 1



- the local stress field has been derived explicitly and has been shown to be singular:

$$\sigma_{ij} = \rho^{\alpha} f_{ij}(\theta,\phi)$$

Results

(i) isotropic laminae (limit case)

α is function of the material constants $C_{ij}^{(m)}$ and independent of θ. Moreover it is exactly the same as that predicted by the 2D theory (Bogy).

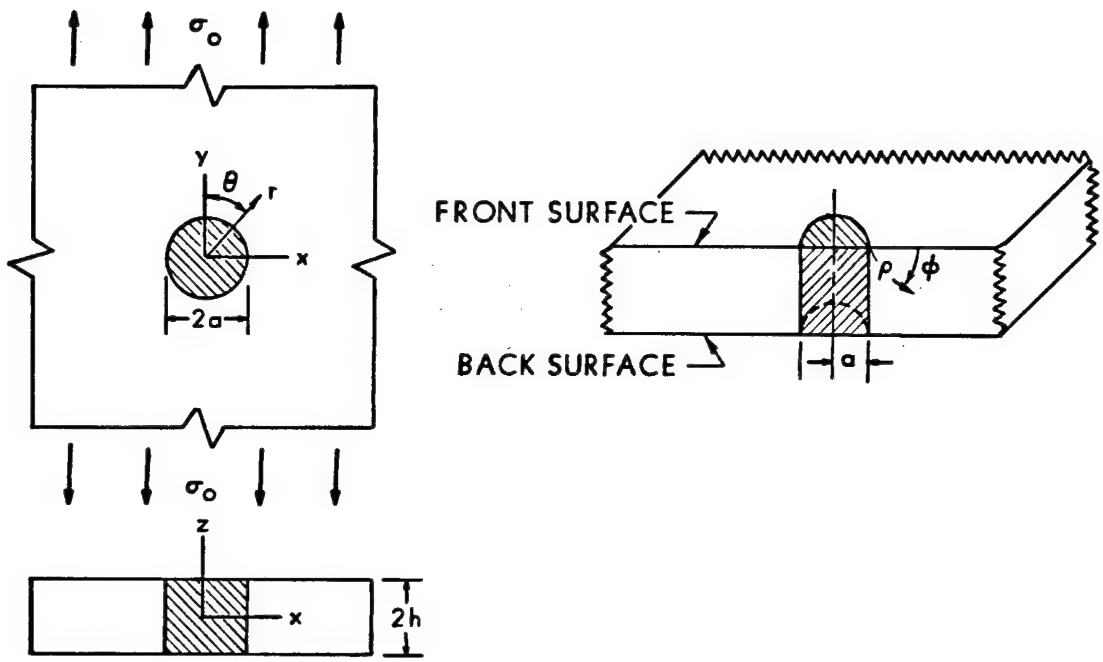
(ii) transversely isotropic laminae

α is a complicated function of the material constants $C_{ij}^{(m)}$ as well as the angular distribution θ.

- The interlaminar stresses can now be used to predict possible failures due to debonding at the interface and adjacent to the hole (this location is the most critical).

CYLINDRICAL INCLUSION EMBEDDED IN A PLATE

Model: 3D Elasticity
Materials: Both Plate and Inclusion Homo. and Isotropic
Method of Solution: Analytical
Objective: The 3D stress field in the vicinity of the inclusion. (The inclusion may be thought of as a fiber.)



Results:

- the displacement and stress fields have been recovered. For experimental verification, we plot the displacement u_{zz} at the plate surface $z = h$ and for $\theta = \pi/2$:
- at the intersection of the inclusion and the free surface of the plate, the stress field is singular. The strength of the singularity is exactly the same as that predicted by the 2D theory (Bogy).

Extention: The consideration of an additional load (tensile or compressive) along the axis of the fiber.

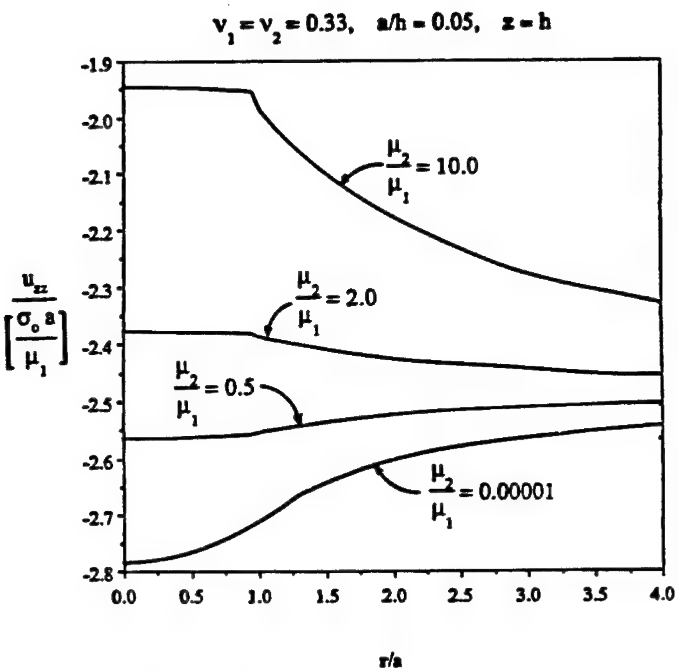


Figure 14. Displacement u_{zz} as a function of r at z=h, $\theta=\pi/2$ for a/h=0.05, $\nu_1=\nu_2=0.33$ and different values of μ_2/μ_1.

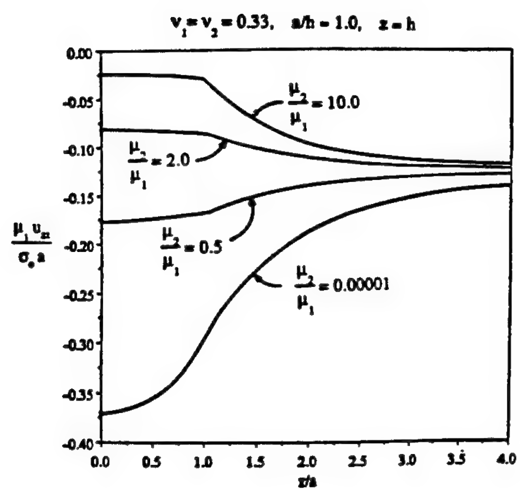
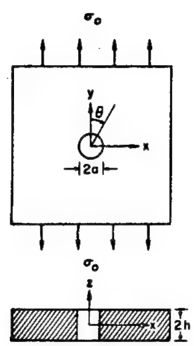


Figure 15. Displacement u_{zz} as a function of r at z=h, $\theta=\pi/2$ for a/h=1.0, $\nu_1=\nu_2=0.33$ and different values of μ_2/μ_1.

3D STRESS FIELD SINGLE LAYER (ISOTROPIC MATERIAL)

Model: 3D, linear elasticity
Material: Homogeneous and Isotropic
Method of Solution: Analytical



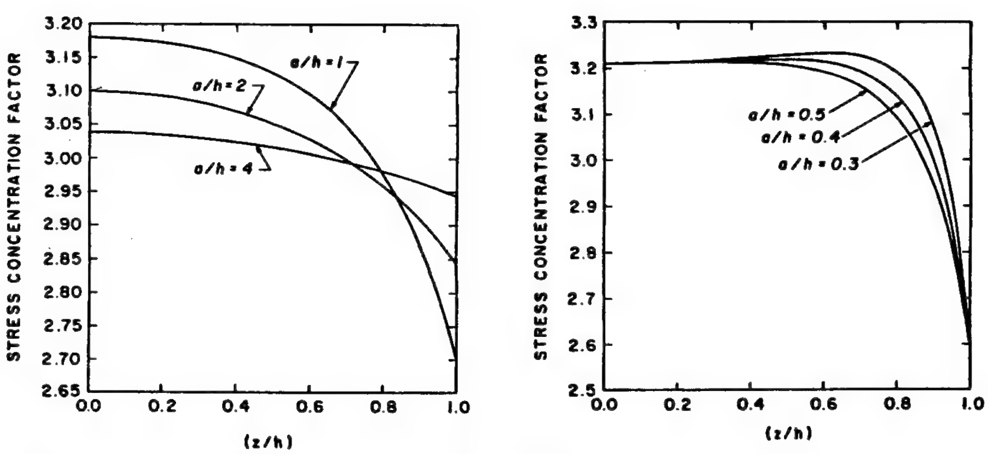
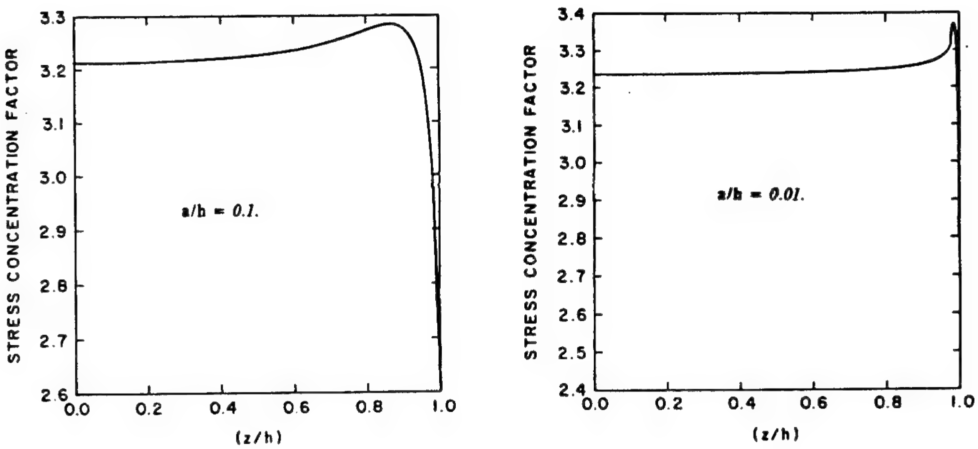


FIG. 2—*Stress concentration factor across the thickness for Poisson's ratio ν = 0.33 and various a/h ratios.*



REMARKS: the analysis shows that:

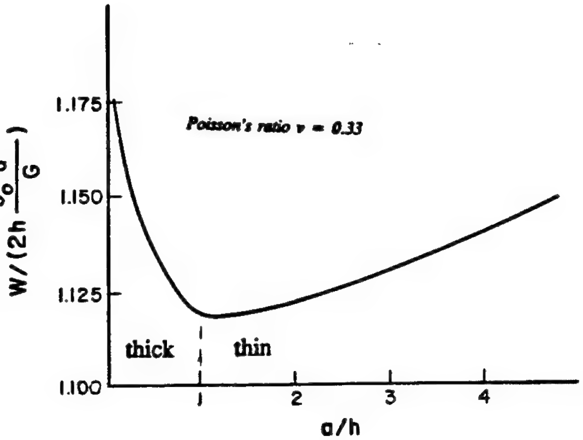
- for ratios of (a/h) > 0.5 the max. s.c.f. (stress concentration factor) occurs at the middle plane which implies that cracks are more likely to initiate at the center of the plate

- for ratio s of (a/h) < 0.5 the max. s.c.f. occurs close to the free surface (in fact, one radius distance away from the surface) which implies that cracks are most likely to initiate close to the free surface

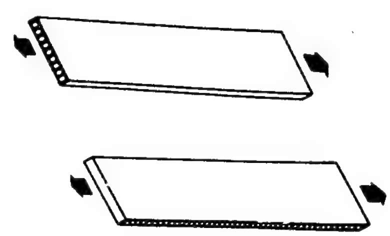
- for very large (a/h), i.e. $4 < a/h < \infty$, the plane stress solution gives very good approximations (error is less than 1%)

- for relatively thick plates (a/h < 0.5), the s.c.f. is slightly higher (by as much as 8%) than that predicted by the plane strain solution (3).

  While this does not have appreciable affect on fast fracture, it does however have a substantial reduction to the fatigue life span of the structure.



1.175
1.150
1.125
1.100
Poisson's ratio ν = 0.33
$W/(2h\frac{\sigma_o^2 a^2}{G})$
thick
thin
1
2
3
4
a/h



Residual Stress Development During Processing of Composite Laminates

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Residual stresses are developed during processing of composite materials due to chemical shrinkage of the matrix and thermal mismatch of the constituent materials. These stresses may be high enough to cause transverse cracking even before any external loading takes place [1]. The push for new, high temperature materials has meant higher processing temperatures. These higher processing temperatures will lead to higher residual stressses. Thus, the need to be able to predict and control residual stresses is becoming increasingly important.

Unfortunately, how these residual stresses develop during processing is not fully understood. Furthermore, to model their development during cure the effect of cure on composite mechanical properties must be understood.

To visualize residual stress development unsymmetric cross-ply laminates were used. These types of laminates will warp after cure. The magnitude of this warping can be related to the magnitude of the residual stresses induced [2]. Specifically, $[0_4/90_4]_T$ laminate strips (1"x6" and 2.5"x8.5") were used. These laminates were cured using compression molding in a computer controlled hot press. At several points during the manufacturer's recommended cure cycle, the process cycle was stopped and the specimens cooled down to room temperature at a fixed rate.

Residual stress development (in the form of dimensionless curvature) has been found to be proportionately related to the degree of cure. Initially the curvature is insignificant, but increases dramatically around the gel time until reaching the fully cured value. A plot of degree of cure versus cure time shows the same characteristics.

It has been found that thermal strains remain relatively constant with cure time (degree of cure). Elastic analysis which neglects chemical shrinkage has predicted the curvature quite well. This indicates that the effect of chemical shrinkage on residual stress may be negligible. Postcure studies on T300/3501-6 support this concept. Transverse tension studies have shown that the transverse modulus shows the same characteristics as dimensionless curvature and degree of cure versus cure time.

Future work on this project will include efforts to optimize the cure cycle to reduce residual stresses both analytically and experimentally. A viscoelastic analysis is to be performed to characterize the stress relaxation which takes place during the cure cycle. Creep studies will be done to provide an experimental data base of relaxation modulus as a function of cure time.

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RESIDUAL STRESS DEVELOPMENT DURING PROCESSING OF COMPOSITE LAMINATES

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OBJECTIVES

- Identify the mechanisms of residual stress development during processing of polymer matrix composites
- Develop a prediction methodology
- Develop a procedure to control residual stresses through process cycle optimization

APPROACH

- Non-Mechanical Strain:

$$e_n = e_c(c) + e_t(T)$$

- Residual Stress:

$$\sigma_R = -\int_0^t E[\xi(t) - \xi(\tau)](\dot{e}_c + \dot{e}_T)d\tau$$

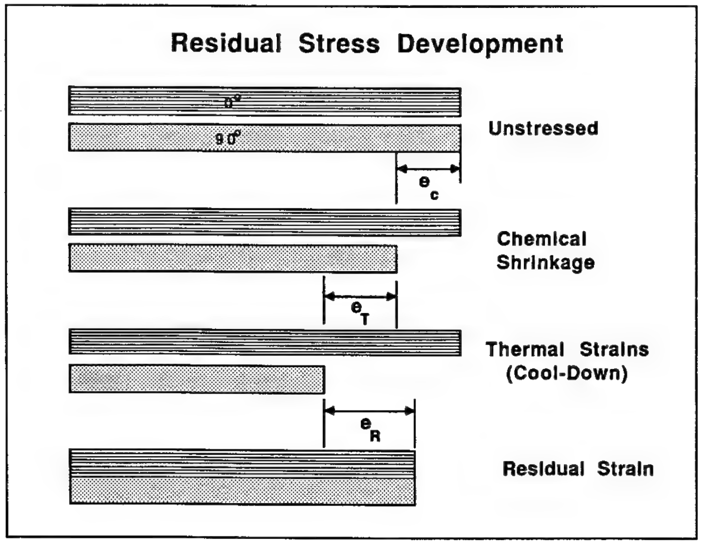
$$\xi(u) = \int_0^u \frac{ds}{a[T(s),c(s)]}$$

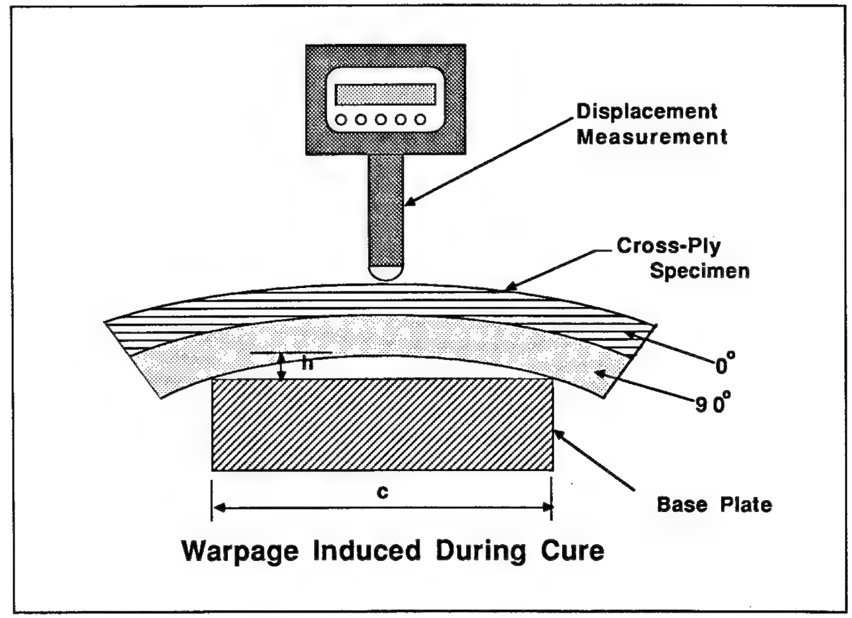
ξ : Reduced Time
a : Shift Factor

- Experimentally determine residual stress development through;

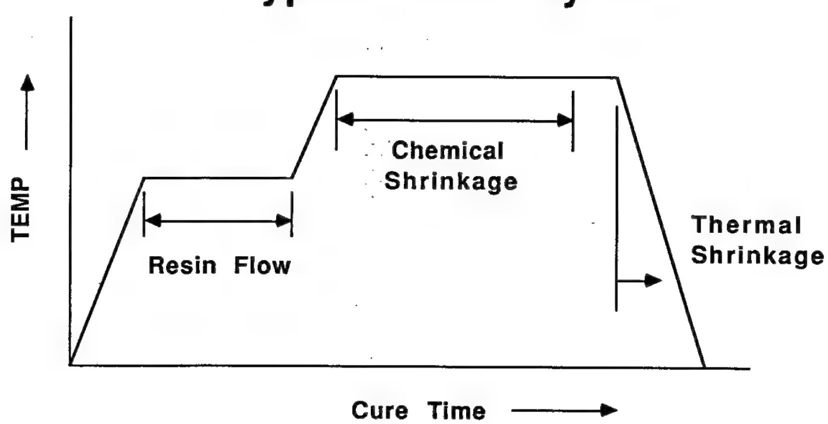
  1) intermittent cure of cross-ply laminates
  2) continuous cure with surface mounted strain gages

- Identify controlling parameters in residual stress development
- Model the cure process and residual stress development through viscoelastic analysis and experimental correlation
- Monitor optimization schemes for mechanical properties (i.e. strength, stiffness, etc.)

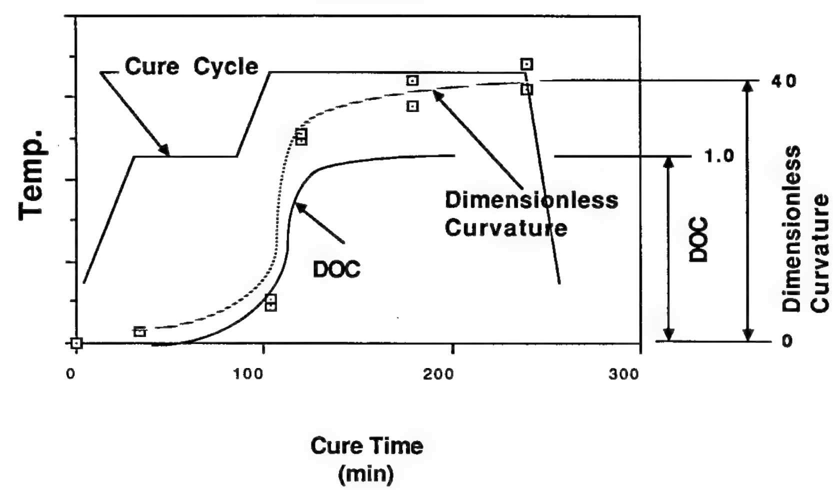


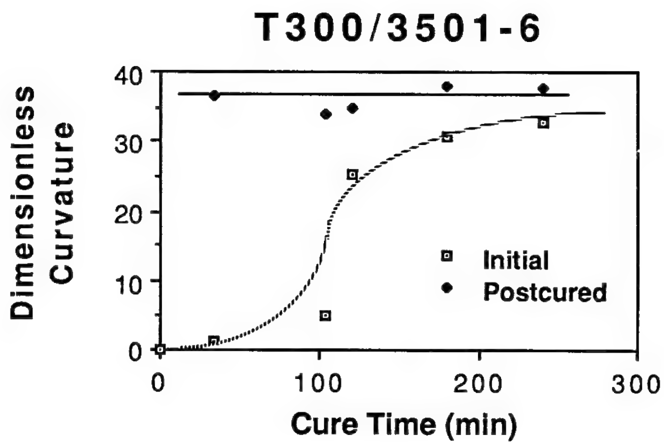


Typical Cure Cycle

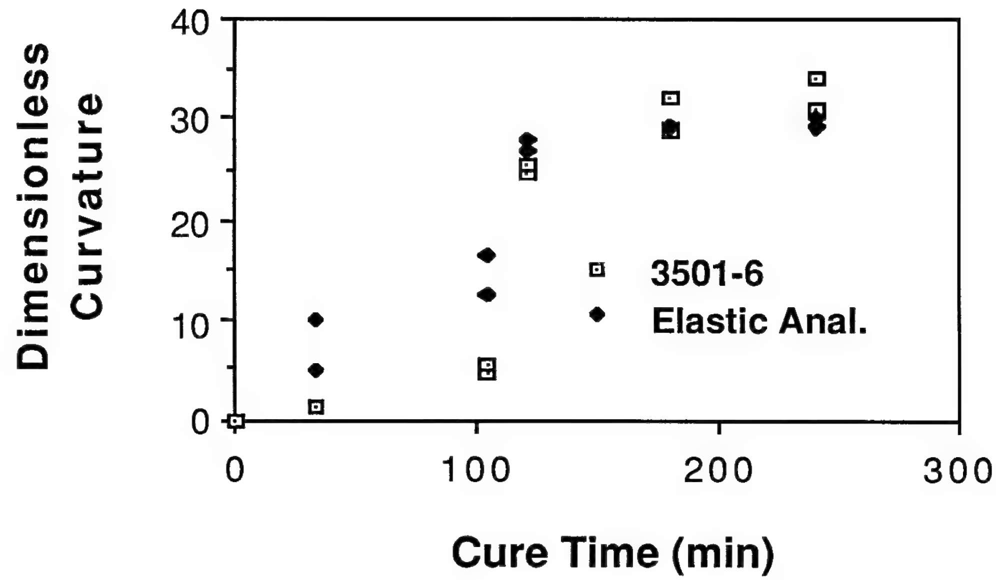


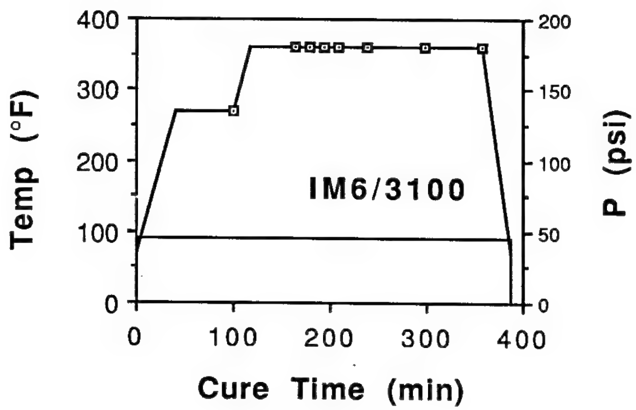
T300/3501-6



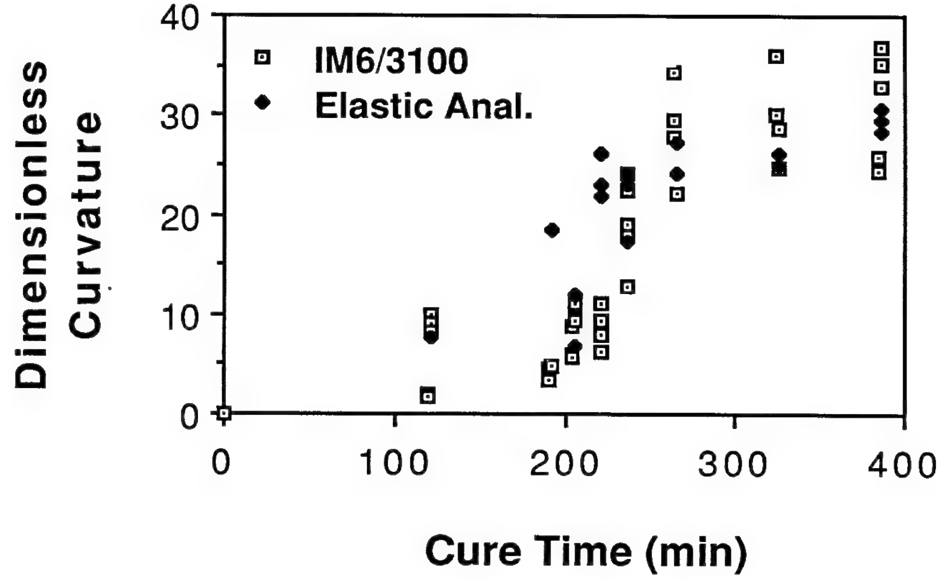


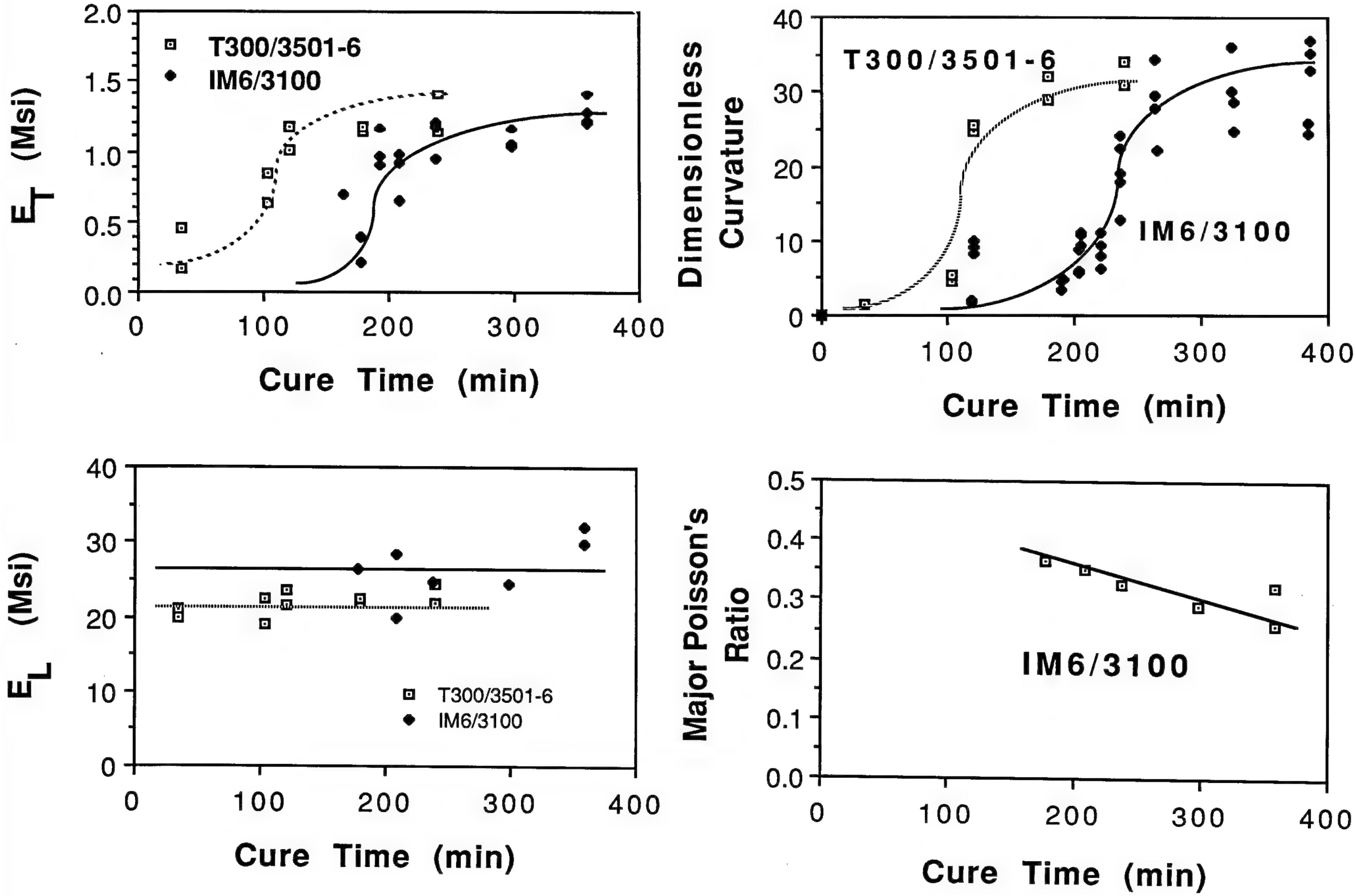
Effect of Postcure on Curvature



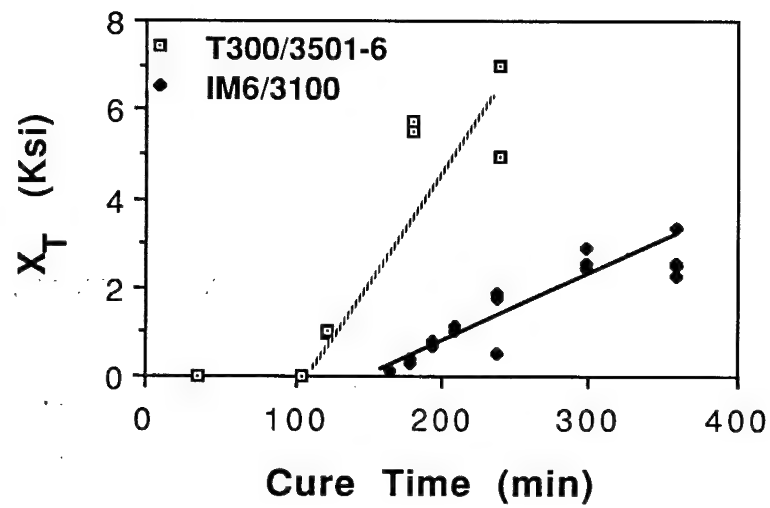


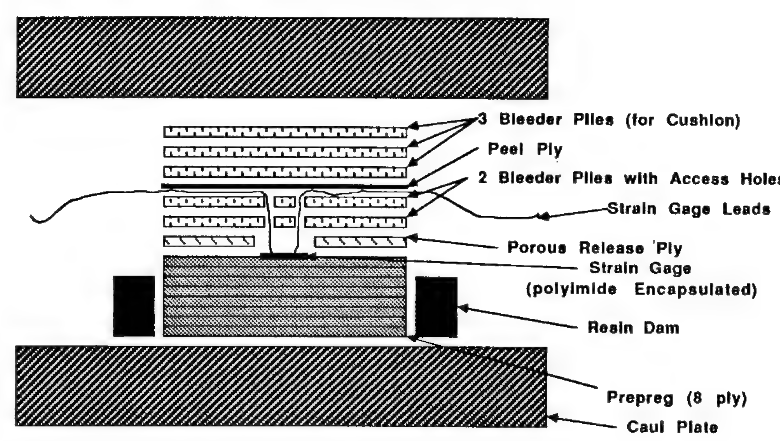
Gr/BMI Cure Cycle



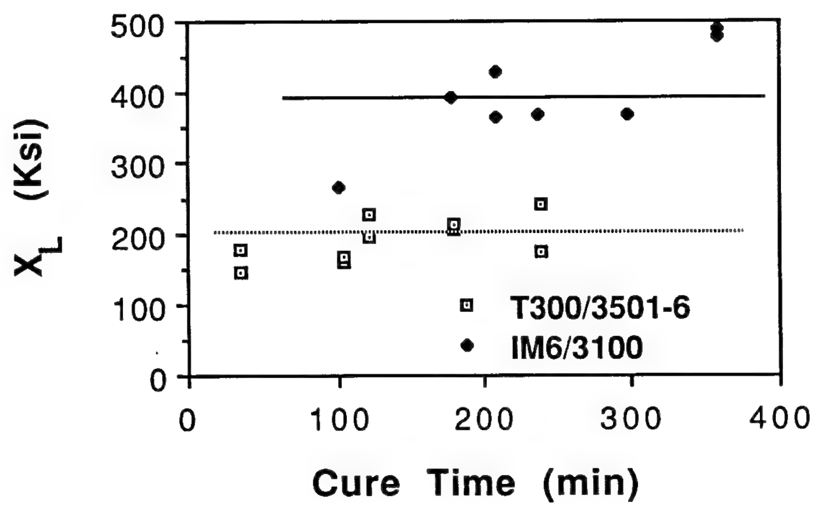


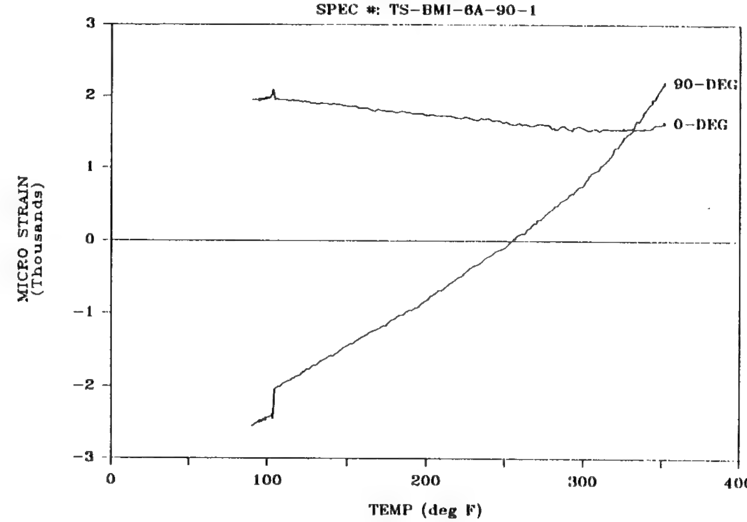
T300/3501-6
IM6/3100
E_T (Msi)
Cure Time (min)
Dimensionless Curvature
E_L (Msi)
Major Poisson's Ratio

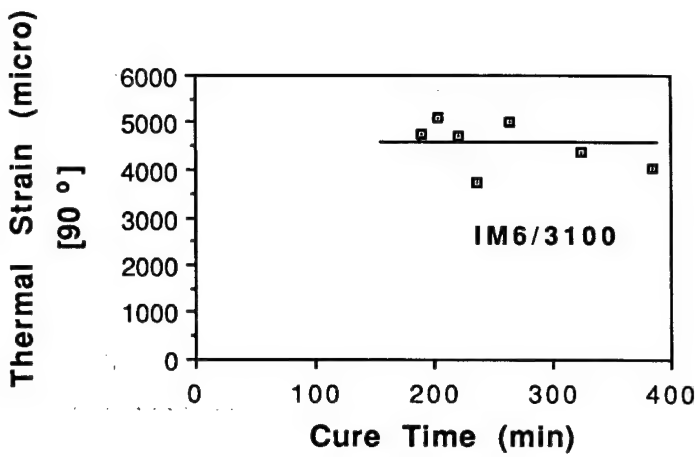




Thermal Strain Experimental Lay-up







Effect of Cure Time on Thermal Strain

SUMMARY

- Thermal strain dependence on state of cure is negligible
- Elastic analysis to date has predicted curvature well indicating that the effect of chemical shrinkage on residual stress is negligible
- Thermal shrinkage is of primary importance in residual stress build-up
- Residual stress (dimensionless curvature), transverse Young"s modulus, and degree of cure change in proportion to each other during the cure cycle

Future Work

- Optimization of cure cyle to reduce residual stress (analytically and experimentally)
- Viscoelastic analysis to predict $E_r(t)$ during the cure cycle
- Creep studies on Gr/BMI for partially cured specimens

EQUILIBRIUM-BASED ITERATIVE STRESS ANALYSIS APPROACHES FOR ADHESIVE JOINTS

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ABSTRACT

In an adhesive joint with a short lap length, it can be assumed that the adherends slide parallel to each other as rigid bodies. Under this assumption the shear stress along the bond line is uniform, since the difference between the adherend displacements which kinematically controls the bond shear strain is uniform. By equilibrium, a linear buildup of axial stress must be then present in the adherends. This simplified viewpoint gives a statically determinate relationship for the maximum bond shear stress which will requires it to be equal to the applied axial stress times the ratio of bond thickness to length, a calculation which has often used for estimating required lap lengths in joint design, although it tends to be on the unsafe side because of shear stress concentrations occurring at the ends of the joint. A first estimate for the required correction can be made by calculating the adherend stretching deformations from the linear adherend axial stress distribution and introducing this into the kinematic relation for the shear strain to get a first correction for the shear stress. It will be found that the correction is significant if the relationship:

$$\frac{E_x}{G_b} \leq \frac{\ell^2}{tt_b}$$

between the bond shear modulus, G_b, adherend axial Young's modulus, E_x, lap length, ℓ, adherend thickness t, and bond thickness, t_b , is applicable. For large values of ℓ, an iterative process in which new updates of the shear stress obtained from correcting the axial deflection estimate are interspersed with improvements of axial stress distribution using the equilibrium relation on a repetitive basis. This can be shown to lead to a Taylor's series approximation for the hyperbolic function expressions for shear and axial stress normally obtained from shear lag models for adhesive joints. The strategy illustrated by this example, ie. starting with a displacement assumption for one component of a system, using equilibrium to get stresses in a second component for which deflections are believed to be negligible, and incorporating these steps in an iterative process to get successive improvements, forms the basis for introducing thick plate effects into joint stress analysis which are the subject of this presentation [1].

Thick plate effects are highly significant in organic composite adherends because of the high level of compliance for transverse shear and thickness normal effects. These were treated by analyses developed by Dickson et al [2] and by Renton and Vinson [3] in the early 70's, using a continuum approach which led to an eighth order system of ordinary differential equations describing various quantities of interest. The method treated here is similar in a number of respects, including reduction of the formulation to a system of ordinary differential equations. The present formulation places particular emphasis on understanding the parametric combinations which determine how significant the effects of transverse shear and thickness normal deflections are for a particular material system. To accomplish this, considerable effort has been exerted to state the formulation in terms of dimensionless parameters which govern thickness effects. Moreover, the analysis is applied to a relatively simple example, that of an adherend bonded to a rigid substrate , providing ease in the interpretation of results. Comparison with finite element results is used to evaluate the level of accuracy provided by the analysis.

In addition to the special emphasis of the present approach on clarifying the effects of adherend thickness, the basis for the development of expression for stresses is somewhat different from that of [2,3]. In the latter, effects of transverse shear deformation were based on the assumption that the transverse shear strain is distributed as a smooth quadratic function through the adherend thickness, and transverse shear stresses were obtained from the constitutive equations. In the present approach, the starting point is the linear axial displacement distribution of classical beam theory together with the corresponding axial stress distribution, from which the transverse

shear and thickness normal stresses are obtained by equilibrium with respect to the axial stress. Transverse shear and thickness normal **strain** corrections then follow, using the constitutive relations, from which a corrected expression for the axial displacement is obtained. At this point the axial stress is recalculated, from which moment and stress resultant are evaluated and inserted into equilibrium equations with respect to the bond shear and peel stress. For cases of homogeneous adherends, it apears that the results obtained from the present approach will be indistinguishable from those using the approach of [2,3]. In the case of laminated adherends, however, significant differences will be found. For example the earlier approach applied to a 0/90 laminate would tend to give a smooth parabolic distribution of transverse shear stress in conjunction with the assumed smooth parabolic shear strain distribution, since the shear modulus would be uniform through the laminate thickness for this case. The present approach, which would predict a stepwise linear axial stress distribution corresponding to standard laminate theory, would lead to stepwise quadratic and cubic distributions of transverse shear and thickness normal stresses, respectively.

As with the shear lag example described at the beginning of this abstract, the process can, in principle, be recycled indefinitely to get further improvements in the calculations. The version treated here is restricted to the first update for practical reasons. Comparison with the FE results suggest that for typical joint parameters, the first correction gives adequate accuracy except at the interior of the bond layer near the ends of the joint, where a stress singularity would be indicated in an elasticity analysis. Conceivably, a special continuum element could be devised for the small region where the singularity predominates and combined with the present approach which appears to give an excellent description of the stresses and deflections within the adherend.

In the presentation, the essential features of the analytical formulation will be described, followed by a comparison of numerical results between the analytical and FE approaches, for the two extremes of adherend transverse stiffness which would normally be encountered, ie. unidirectional graphite epoxy representing the extreme of high shear and thickness normal compliance, vs. aluminum representing the extreme of small compliance for these deformations. Comparison of predicted results for the stresses along the bond/adherend interface for these cases indicates that the analytical results are nearly indistinguishable from the FE results, which, because of careful mesh refinement near the corner regions where high stress gradients occur, are believed to be quite accurate except for very small distances from the corners. Comparisons are also made, for the graphite epoxy case, to predictions of stresses through the adherend thickness, indicating that the particular functional representations used for modelling the thickness-wise variations of stresses are appropriate.

Comments will be made on lower-order versions of the present analysis, some of which give surprising levels of accuracy, especially for the shear stress along the bondline.

The present approach involves functional forms for the thickness effects that are similar to those used in recent variational formulations for thick plates [4]. Future efforts aimed at comparing the two approaches are planned. It is of interest that the present approach can be easily generalized to two in-plane dimensions to allow for semi-three dimensional analyses of edge effects, finite width effects in adhesive joints, thickness effects around cutouts and other problems of interest to be investigated.

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EQUILIBRIUM-BASED ITERATIVE STRESS ANALYSIS APPROACHES FOR ADHESIVE JOINTS

by
D. W. Oplinger

ARMY MATERIALS TECHNOLOGY LABORATORY

Watertown, MA

presented at

THE THIRTEENTH MECHANICS OF COMPOSITES REVIEW

Bal Harbour, Florida
November, 1988

OBJECTIVES

- PROVIDE ANALYTICAL METHOD FOR ELUCIDATING ADHEREND THICKNESS EFFECTS IN ADHESIVE JOINTS CONTAINING COMPOSITE ADHERENDS
- APPLY TO TUTORIAL ON THEORETICAL CONSIDERATIONS FOR JOINTS IN MIL HDBK 17

ANALYTICAL APPROACH FOR AGR2 DEVELOPMENT

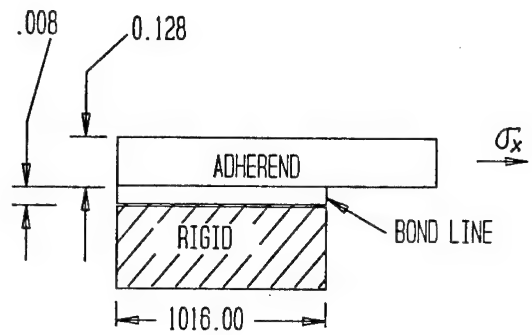
1.) ASSUME AXIAL STRESSES CORRESPONDING TO STRETCHING PLUS CLASSICAL BEAM BENDING IN THE ADHEREND.

2.) USE STRESS EQUILIBRIUM EQUATIONS TO CALCULATE TRANSVERSE SHEAR AND THICKNESS NORMAL STRESSES FROM THE AXIAL STRESSES.

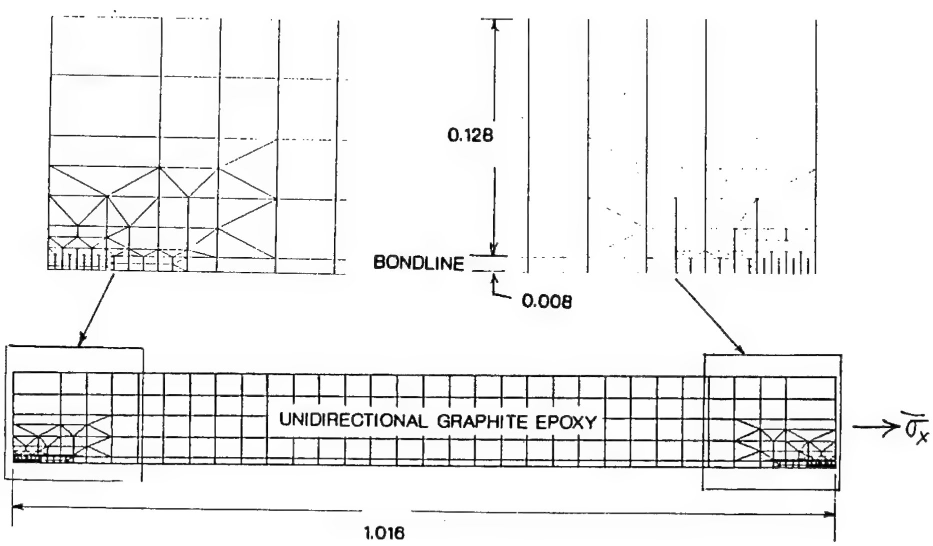
3.) INTEGRATE THE STRAINS OBTAINED FROM THE STRESSES PROVIDED IN STEP 2 THROUGH THE ADHEREND THICKNESS TO GET DISPLACEMENT CORRECTIONS.

4.) USING THE CORRECTED DISPLACEMENTS, FORM UPDATED AXIAL STRESS, STRESS RESULTANT AND MOMENT EXPRESSIONS FOR THE ADHERENDS.

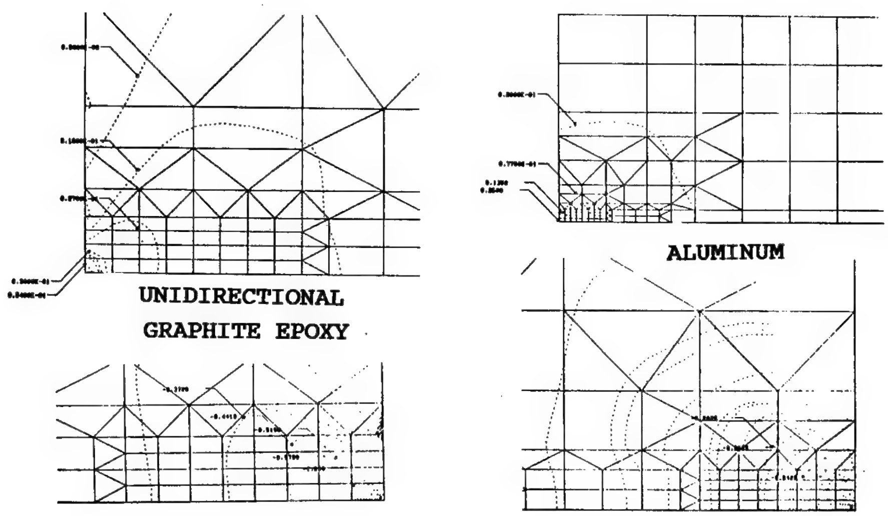
5.) INSERT THE UPDATED MOMENT AND STRESS RESULTANT INTO INTEGRATED EQUILIBRIUM EXPRESSIONS RELATING THEM TO THE ADHESIVE SHEAR AND PEEL STRESS, TO GET A SYSTEM OF ORDINARY DIFFERENTIAL EQUATIONS WHICH CAN BE SOLVED BY CLASSICAL METHODS.



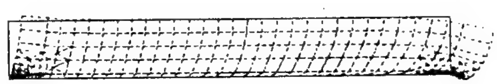
GEOMETRY OF INTEREST -- BONDED STRIP ON RIGID SUBSTRAT



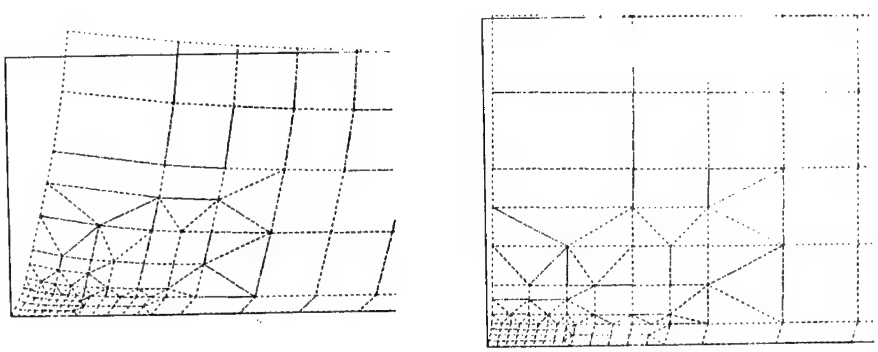
FINITE ELEMENT MODEL



σ_z STRESS CONTOURS FROM FE SOLUTION



CHARACTERISTIC DISTORTED SHAPES FROM FE SOLUTION



UNIDIRECTIONAL GRAPHITE EPOXY

ALUMINUM

NOTATION

MATERIAL PROPERTIES

Adherend

E_x, E_z, G_{xz} -- axial Young's modulus, transverse Young's modulus and transverse shear modulus

Adhesive

E_b, G_b -- Young's modulus, shear modulus

DIMENSIONAL VARIABLES

t, t_b -- adherend, bond layer thickness

ℓ -- length of overlap

STRESSES

ADHEREND	ADHESIVE
σ_x ≡ axial normal stress	$\sigma \equiv \sigma_z$ in bond layer
σ_z ≡ transverse " "	$\tau \equiv \tau_{xz}$ " " "
τ_{xz} ≡ " shear "	

DISPLACEMENTS

u, w -- AXIAL, TRANSVERSE DISPLACEMENT

u_0, w_0 -- AT ADHEREND/BOND INTERFACE

DIMENSIONLESS RATIOS

$$\rho_G = \frac{E_x}{G_{xz}} \qquad \rho_E = \frac{E_x}{E_z} \qquad \rho_{Gb} = \frac{E_x}{G_b} \qquad \rho_{Eb} = \frac{E_x}{E_b}$$

$$\rho_t = \frac{t_b}{t} \qquad \lambda = \frac{\ell}{t} \qquad \xi = \frac{x}{t} \qquad \zeta = \frac{z}{t}$$

$$R_1 = \rho_{Gb}\rho_t \qquad R_2 = \rho_{Eb}\rho_t$$

DERIVATIVE NOTATION

$$d_k f(\xi) = \frac{d^k f}{d\xi^k}$$

ADHEREND MOMENTS AND STRESS RESULTANTS

AXIAL STRESS RESULTANT	MOMENTS ABOUT ADHEREND NA	MOMENTS ABOUT BONDLINE
$N_x = t\int_0^1 \sigma_x \, d\zeta$	$M_x = t^2\int_0^1 (\frac{1}{2} - \zeta)\sigma_x \, d\zeta$	$M_{x0} \equiv t^2\int_0^1 -\zeta^2\sigma_x \, d\zeta$

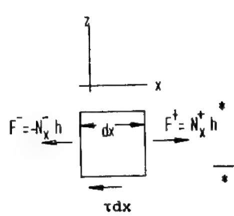
DIMENSIONALIZED (STRESS-LIKE) STRESS RESULTANTS

$$n_x = \frac{N_x}{t} \quad ; \quad m_x = \frac{M_x}{t^2} \quad ; \quad m_{x0} = \frac{M_{x0}}{t^2}$$

(A) AXIAL FORCE BALANCE

$$\sum F_x = (N_x^+ - N_x^-)h - \tau dx = 0$$

$$\frac{\partial N_x}{\partial x} - \tau = 0$$

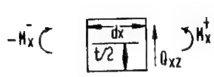


* h = DEPTH NORMAL TO VIEW

(B) MOMENT BALANCE

$$\sum M = (M_x^+ - M_x^-)h + Q_{xz}hdx - \tau\frac{t}{2}hdx$$

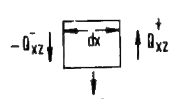
$$\therefore \frac{\partial M_x}{\partial x} + Q_{xz} - \tau\frac{t}{2} = 0$$



(C) TRANSVERSE FORCE BALANCE

$$\sum F_z = (Q_{xz}^+ - Q_{xz}^-)h - \sigma hdx$$

$$\therefore \frac{\partial Q_{xz}}{\partial x} - \sigma = 0$$



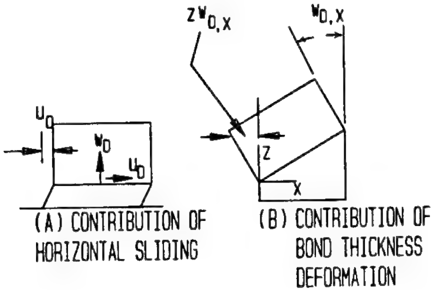
ADHEREND-ADHESIVE EQUILIBRIUM RELATIONS

$$N_{x,x} = \tau \qquad \frac{t}{2}\tau - M_{x,x} = Q_{xz} \qquad Q_{xz,x} = \sigma$$

(alternatively: $M_{x0,x} = -Q_{xz}$)

IN TERMS OF n_x, m_{x0} :

$$d_1 n_x = \tau \qquad d_2 m_{x0} = -\sigma$$



ADHESIVE STRESSES VS. DISPLACEMENTS

$$u_0 = \frac{t_b}{G_b}\tau - \frac{t_b^2}{E_b}\sigma_{,x} \qquad w_0 = \frac{t_b}{E_b}\sigma$$

DEFORMATIONS ASSOCIATED WITH BONDLINE

INITIAL ADHEREND DISPLACEMENTS AND STRESSES

DISPLACEMENT

$$u^\circ(z) = u_0 - \frac{t}{2} w_{0,x} + (\frac{t}{2} - z)\, w_{0,x}$$

STRESS

$$\sigma_x^\circ = E_x u^\circ_{,x}$$

(IN TERMS OF MOMENT AND STRESS RESULTANT)

$$\sigma_x^\circ = \frac{N_x^\circ}{t} + 12\frac{M_x^\circ}{t^2}(\frac{1}{2} - \zeta)$$

(IN TERMS OF DIMENSIONLESS PARAMETERS)

$$\sigma_x^\circ = R_1 d_1 \tau - R_2(\rho_t + \zeta) d_2 \sigma$$

MOMENTS AND STRESS RESULTANT

$$n_x^\circ = R_1 d_1 \tau - R_2 d_2 \sigma \;; \quad m_x^\circ = \frac{1}{12} R_2 d_2 \sigma \quad m_{x0}^\circ = -\frac{n_x^\circ}{2} + \frac{1}{3} R_2 d_2$$

STRESS-EQUILIBRIUM AND TRACTION RELATIONS

$$\sigma^\circ_{x,x} = -\tau^\circ_{xz,z} \;; \quad \tau^\circ_{xz,x} = -\sigma^\circ_{z,z}$$

$z = t$:

$$\tau^\circ_{xz})_{z=t} = \sigma^\circ_z)_{z=t} = 0$$

SATISFIED BY:

$$\tau^\circ_{xz}(\zeta) = \int_\zeta^1 \frac{\partial \sigma_x(\zeta')}{\partial x} d\zeta' \;; \quad \sigma^\circ_z(\zeta) = \int_\zeta^1 \frac{\partial \tau_{xz}(\zeta')}{\partial x} d\zeta'$$

UPDATED EXPRESSIONS

$$\varepsilon_z = \frac{\sigma_z}{E_z} \;; \quad \gamma_{xz} = \frac{\tau_{xz}}{G_{xz}}$$

$$\varepsilon_z = w_{,z} \;; \quad \gamma_{xz} = u_{,z} + w_{,x}$$

$$w^1 = w_0 + \int_0^\zeta \frac{\sigma_z}{E_z} d\zeta'$$

$$u^1 = u_0 + \int_0^\zeta (\frac{\tau_{xz}}{G_{xz}} - w_{,x})\, d\zeta'$$

$$\sigma_x^1 = E_x u^1_{,x}$$

SPECIAL FUNCTIONS OF ζ

	k=1	k=2	k=3
D_k	1	$\frac{1}{2} - \zeta$	---
E_k	$1 - \zeta$	$6(\zeta^2 - \zeta)$	$1 - 4\zeta + 3\zeta^2$
F_k	$\frac{1}{2} - \zeta + \frac{\zeta^2}{2}$	$-\frac{1}{12} + \frac{\zeta^2}{4} - \frac{\zeta^3}{6}$	$-\zeta + 2\zeta^2 - \zeta^3$
f_k	$\zeta - \frac{\zeta^2}{2}$	$-\frac{\zeta^2}{4} + \frac{\zeta^3}{6}$	$\zeta - 2\zeta^2 + 2\zeta^3$
g_k	$\frac{\zeta^2}{2} - \frac{\zeta^3}{6}$	$-\frac{\zeta^3}{12} + \frac{\zeta^4}{24}$	$-\frac{1}{2}\zeta^2 + \frac{2}{3}\zeta^2 - \frac{1}{4}\zeta^4$
h_k	$\frac{\zeta^3}{6} - \frac{\zeta^4}{24}$	$-\frac{\zeta^4}{48} + \frac{\zeta^5}{120}$	$-\frac{1}{6}\zeta^3 + \frac{1}{6}\zeta^4 - \frac{1}{20}\zeta^5$

REQUIRED DEFINITE INTEGRALS

$$I_1 \equiv \int_0^1 f_3 d\zeta \;; \quad I_2 \equiv \int_0^1 f_2 d\zeta \;; \quad I_3 \equiv \int_0^1 h_3 d\zeta \;; \quad I_4 \equiv \int_0^1 h_2 d\zeta$$

$$J_1 \equiv \int_0^1 f_3 \zeta d\zeta \;; \quad J_2 \equiv \int_0^1 f_2 \zeta d\zeta \;; \quad J_3 \equiv \int_0^1 h_3 \zeta d\zeta \;; \quad J_4 \equiv \int_0^1 h_2 \zeta d\zeta$$

EVALUATION

I_1	I_2	I_3	I_4
$\frac{1}{12}$	$-\frac{1}{2}$	$-\frac{1}{60}$	$-\frac{2}{15}$
J_1	J_2	J_3	J_4
$\frac{1}{30}$	$-\frac{7}{20}$	-0.012698	-0.097619

EXPLICTIC EXPRESSIONS:

$$\tau^0_{xz} = E_1\tau - E_2\frac{M^0_{x,x}}{t}$$

$$\sigma^0_z = F_3 d_1\tau - F_2\sigma$$

$$w^1 = w_0 + \frac{t}{E_z} g_3 d_1\tau - \frac{1}{E_z} g_2 \sigma$$

$$w^1_{,x} = w_{0,x} + \frac{t}{E_z} g_3 d_2\tau - \frac{1}{E_z} g_2 d_1\sigma$$

$$u^1_{,z} = - w_{0,x} + E_1\frac{\tau}{G_{xz}} + \frac{E_2}{t}\frac{M^0_{x,x}}{G_{xz}} - \frac{t}{E_z} h_3 d_2\tau + \frac{t}{E_z} h_2 d_1\sigma$$

$$u^1 = u^0 + tf_1\frac{\tau}{G_{xz}} + f_2\frac{M^0_{x,x}}{G_{xz}} - \frac{t}{E_z} h_3 d_2\tau + \frac{t}{E_z} h_2 d_1\sigma$$

$$\sigma^1_x = \sigma^0_x + \sigma_{x_1}$$

WHERE

$$\sigma_{x_1} = \rho_G f_3 d_1\tau - \rho_G f_2\sigma - \rho_E h_3 d_2\tau + \rho_E h_2 d_2\sigma$$

UPDATED MOMENT AND STRESS RESULTANT

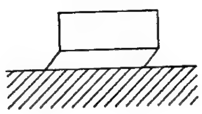
$$n^1_x = (R_1 + \rho_G I_1) d_1\tau - \rho_G I_3 d_3\tau - \rho_G I_2\sigma + [-R_2(\tfrac{1}{2} + \rho_t) + \rho_E I_4)] d_2\sigma$$

$$m^1_{x0} = - (\tfrac{1}{2}R_1 + \rho_G J_1) d_1\tau + \rho_E J_3 d_3\tau + \rho_G J_2\sigma - [\rho_E J_4 - R_2(\rho_t + \tfrac{1}{3})] d_2\sigma$$

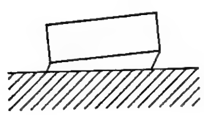
DIFFERENTIAL EQUATIONS

$$\tau = (R_1 + \rho_G I_1) d_2\tau - \rho_G I_3 d_4\tau - \rho_G I_2 d_1\sigma + [- R_2(\tfrac{1}{2} + \rho_t) + \rho_E I_4] d_3\sigma$$

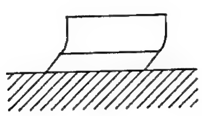
$$\sigma = (\tfrac{1}{2}R_1 + \rho_G J_1) d_3\tau - \rho_E J_3 d_5\tau - \rho_G J_2 d_2\sigma + [\rho_E J_4 - R_2(\rho_t + \tfrac{1}{3})] d_4\sigma$$



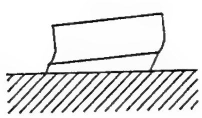
1A) VOLKERSEN SHEAR LAG: $\boxed{\text{VOLK}}$
ADHESIVE SHEAR DEFORMATIONS ONLY



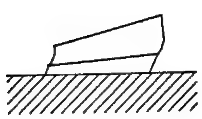
1B) GOLAND REISSNER $\boxed{\text{GR}}$
1. EQUIV. TO VOLKERSEN + BEAM ON ELASTIC FOUNDATION
2. ADHESIVE SHEAR + THICKNESS DEFORMATIONS



1C) ADVANCED VOLKERSEN $\boxed{\text{ADVOLK}}$
VOLKERSEN + ADHEREND SHEAR DEFORMATIONS



1D) ADVANCED GR#1 $\boxed{\text{AGR1}}$
GOLAND REISSNER + ADHEREND
SHEAR DEFORMATIONS

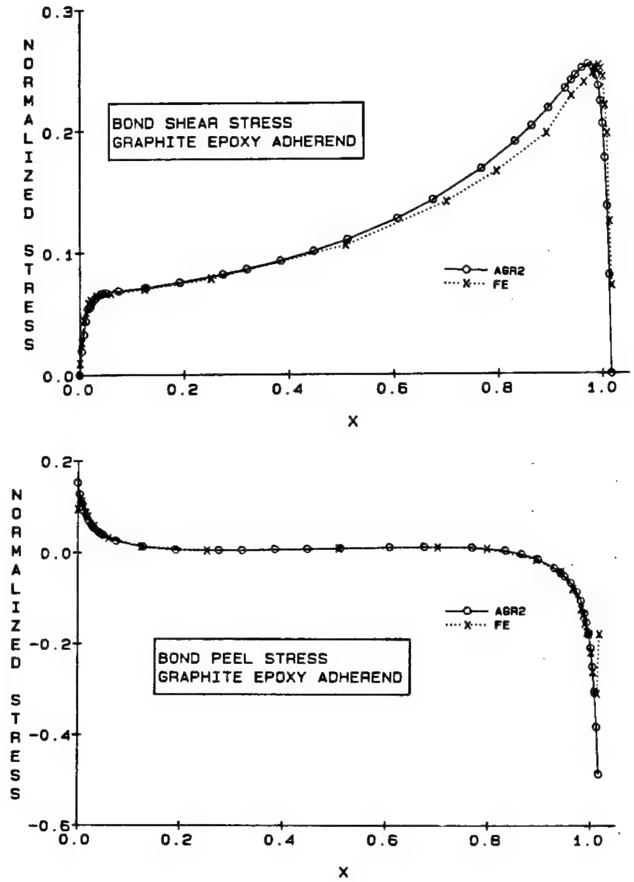


1E) ADVANCED GR#2 $\boxed{\text{AGR2}}$
GOLAND REISSNER + ADHEREND
SHEAR + THICKNESS DEFORMATIONS

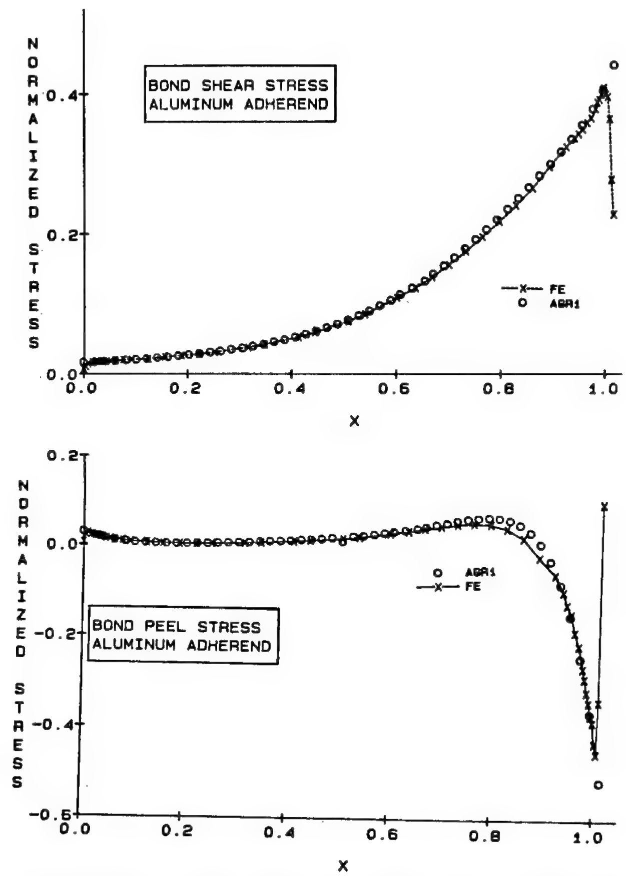
DEFORMATIONS OF INTEREST IN ANALYTIC MODELS

LOWER ORDER SYSTEMS

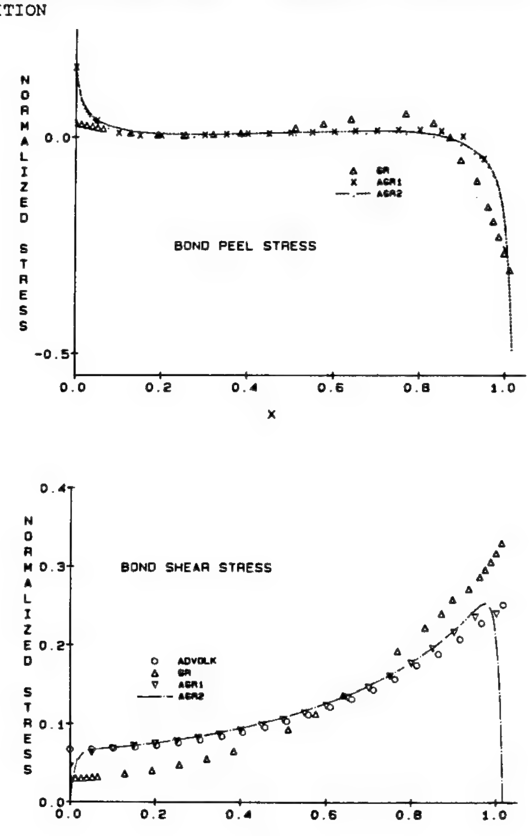
SYSTEM	DELETED PARAMETERS	RETAINED PARAMETERS	ORDER OF SYSTEM (NO. OF ROOTS)	NO. OF BOUNDARY CONDITIONS EACH END
VOLK	ρ_E, ρ_G, R_2	R_1	2	1
ADVOLK	ρ_E, R_2	ρ_G, R_1	2	1
GR	ρ_G, ρ_E	R_1, R_2	6	3
AGR1	ρ_E	ρ_G, R_1, R_2	6	3
AGR2	---	ρ_G, ρ_E, R_1, R_2	8	4



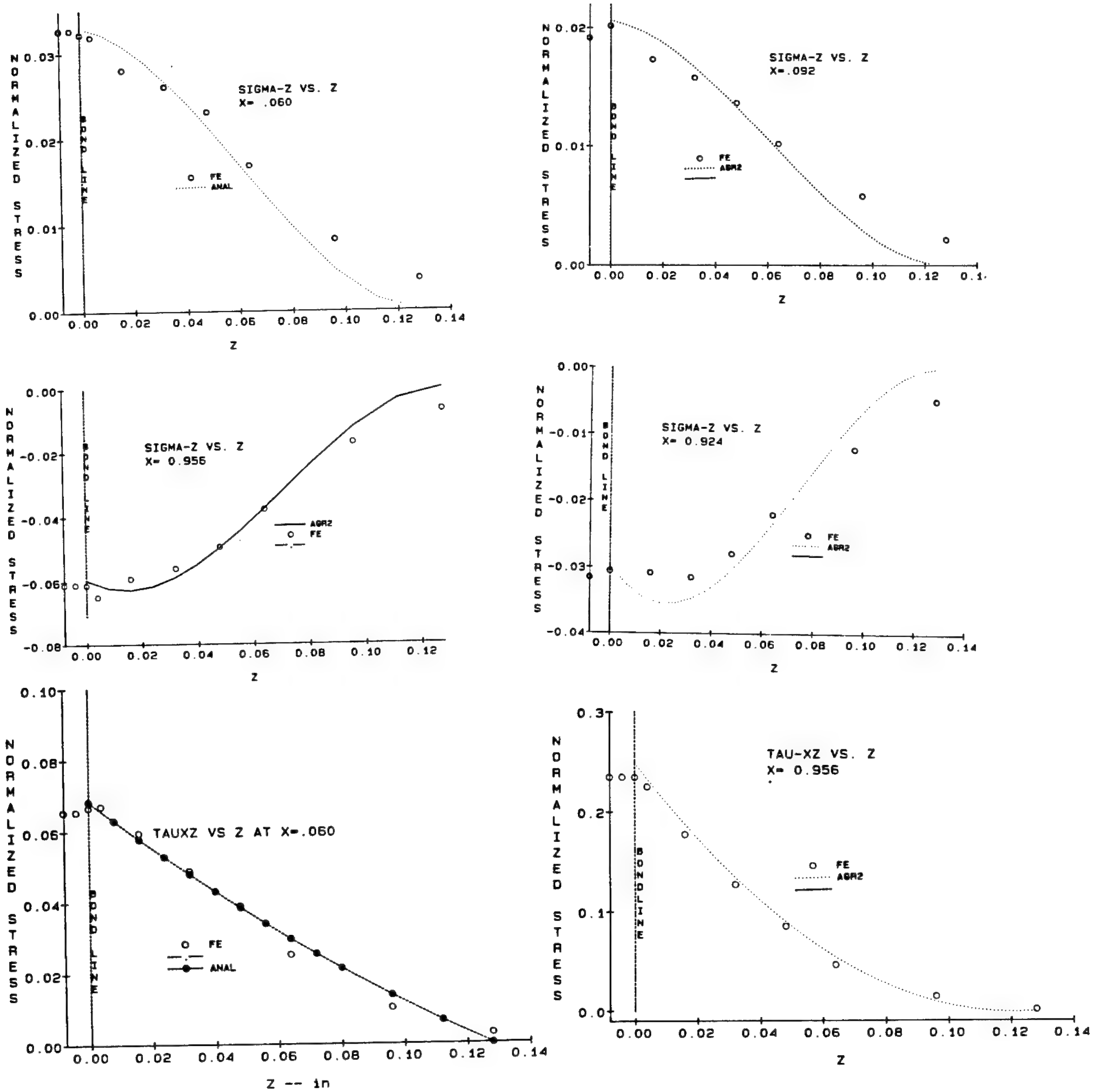
COMPARISON OF ANALYTICAL AND FE RESULTS FOR UNIDIRECTIONAL GRAPHITE EPOXY ADHERENDS -- STRESSES AT INTERFACE VS. AXIAL POSITION



COMPARISON OF ANALYTICAL AND FE RESULTS FOR ALUMINUM ADHEREND -- STRESSES VS. AXIAL POSITION ALONG INTERFACE



COMPARISON OF ANALYTICAL MODELS FOR UNIDIRECTIONAL GRAPHITE EPOXY ADHEREND -- STRESSES VS. AXIAL POSITION ALONG INTERFACE



COMPARISONS OF σ_z AND τ_{xz} VS. z -- UNIDIRECTIONAL GRAPHITE EPOXY

<u>CONCLUSIONS</u>

- THE FIRST ORDER CORRECTION FOR THICK PLATE EFFECTS PROVIDED BY THE ITERATIVE APPROACH OF THIS METHOD PROVIDES EXCELLENT STRESS PREDICITONS FOR TYPICAL ADHEREND MATERIAL SYSTEMS AND GEOMETRIES
- THE PRESENT APPROACH CAN BE GENERALIZED TO 2D PLATE PROBLEMS FOR DEALING WITH FINITE WIDTH EFFECTS IN ADHESIVE JOINTS, AS WELL AS PLATE EDGE EFECTS, EDGE EFFECTS AROUND CUTOUTS, ETC. <u>APPROXIMATE 3D RESULTS WILL BE OBTAINED</u>
- THE PRESENT FORMULATION SHOULD BE COMPARED WITH RECENT VARIATIONAL APPROACHES[4] GIVING SIMILAR THROUGH-THICKNESS STRESS VARIATIONS

AN IMPROVED HIGHER-ORDER THEORY FOR ORTHOTROPIC PLATES

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ABSTRACT

The inherent compliancy of composite laminates in the transverse shear and transverse normal modes of deformation render the classical (Kirchhoff) bending theory, which neglects these effects, inadequate for the purpose of composite stress analysis. Instead, refined bending theories [1-6], admitting transverse shear deformations, are commonly employed. With the exception of Reissner's theory [1-3], refined bending theories [4-6] neglect the transverse normal deformation, thus limiting the range of applicability to the 'moderately' thick regime. In recent years, refined theories have attracted particular attention in the finite element arena, owing to their lower-order (C^0) continuity requirement for the displacement variables -- the aspect allowing generation of simple and efficient models [15-17].

The higher-order bending theories [7-13], which include all modes of deformation, found particular applicability in design-critical regions such as near holes and cutouts, bolted and adhesive joints and laminate edges, where transverse normal deformations can be significant. Insightful reviews of many notable works on this subject can be found, for example, in papers by Lo, Christensen and Wu [13] and Reissner [14].

In this paper we present an improved higher-order displacement theory for orthotropic plates motivated by a finite element perspective, where a simple and variationally sound displacement theory is commonly desired, possessing at most C^0-continuous displacement fields which have the least number of variables (for a detailed discussion of the present theory the interested reader is referred to Tessler [18]). These requirements point toward the lowest order displacement expansions across the thickness suggested by Hildebrand, Reissner and Thomas [7] in the context of a higher-order shell theory; these are linear inplane displacements combined with a parabolic transverse displacement. To simplify explicit integration across the thickness, we expand the displacement components in the thickness direction by means of Legendre polynomials, where Reissner's [1-3] weighted average displacements and two 'auxiliary' transverse displacements are employed. A novel approach is then introduced which eradicates a particular type of shear strain inconsistency -- the conceptual pitfall which has been overlooked in previous investigations [7-10]. The field-inconsistent shear strains, arising from the assumed through-the-thickness displacement distributions, are abandoned in favor of their field-consistent counterparts. The latter strains are equivalent to the former in the mean, and they satisfy shear traction-free boundary conditions at the top and bottom plate faces. In a similar fashion, a cubic variation of the transverse normal strain is introduced into the theory. The principle of virtual work is employed yielding seven partial differential equations of equilibrium, in which two 'auxiliary' displacements appear in their functional form without spatial derivatives; the principle also yields appropriate natural boundary conditions at the plate edges -- five stress resultant conditions and five weighted average displacement conditions. The readily obtained solutions for the 'auxiliary' displacements produce a set of five second-order partial differential equations of equilibrium in terms of five weighted average displacements, thus constituting a 10th-order theory. A close relation of this higher-order theory to Reissner's refined theory [1-3] is identified. To ascertain the level of accuracy attainable with this theory, an analytic solution is obtained for an infinite plate under a sinusoidal normal pressure and compared with the exact elasticity solution and several other plate theories.

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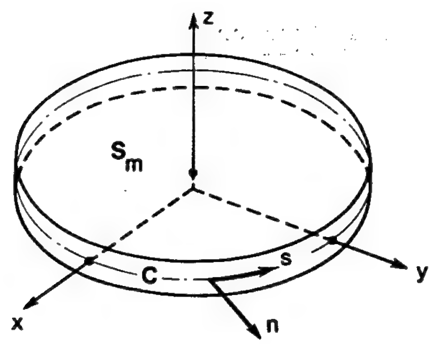
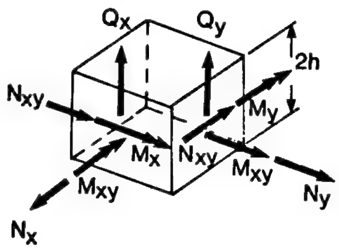
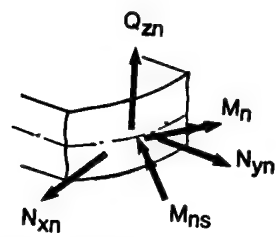


Plate coordinate system



Stress resultants
and moments



Boundary forces
and moments

KINEMATIC ASSUMPTIONS

$$\left.\begin{aligned} u_x(x,y,z) &= u(x,y) + hP_1(\xi)\theta_y(x,y) \\ u_y(x,y,z) &= v(x,y) + hP_1(\xi)\theta_x(x,y) \\ u_z(x,y,z) &= w(x,y) + P_1(\xi)w_1(x,y) \\ &\quad + [P_0(\xi)/5+P_2(\xi)]w_2(x,y) \end{aligned}\right] \quad (1)$$

$P_n(\xi)$ are Legendre polynomials:

$$P_n(\xi) = \frac{1}{2^n\, n!}\, d^n[(\xi^2-1)^n]/d\xi^n \qquad (1.1)$$

$\xi = z/h\ \varepsilon[-1,1]$

$(P_0 = 1,\quad P_1 = \xi,\quad P_2 = (3\xi^2-1)/2)$

Orthogonality Property:

$$\int_{-1}^{1} P_m P_n\, d\xi = \begin{cases} 0 & \text{if } m \neq n \\ 2/(2m+1) & \text{if } m=n \end{cases} \qquad (1.2)$$

WEIGHTED AVERAGE VARIABLES

$$\left.\begin{aligned} u &= \frac{1}{2h}\int_{-h}^{h} u_x dz, \qquad v = \frac{1}{2h}\int_{-h}^{h} u_y dz \\ \theta_x &= \frac{3}{2h^3}\int_{-h}^{h} u_y z\, dz, \qquad \theta_y = \frac{3}{2h^3}\int_{-h}^{h} u_x z\, dz \\ w &= \frac{1}{2h}\int_{-h}^{h} u_z(P_0 - P_2)\, dz \end{aligned}\right] \quad (2)$$

STRAIN-DISPLACEMENT RELATIONS

$$\left.\begin{aligned} \varepsilon_x &= u,_x + hP_1(\xi)\theta_{y,x} \\ \varepsilon_y &= v,_y + hP_1(\xi)\theta_{x,y} \\ \gamma_{xy} &= u,_y + v,_x + hP_1(\xi)(\theta_{y,y} + \theta_{x,x}) \end{aligned}\right] \quad (3)$$

$$\left.\begin{aligned} \gamma_{xz} &= (\theta_y + w,_x) \\ &\quad + P_1(\xi)w_{1,x} + [P_0/5+P_2(\xi)]w_{2,x} \\ \gamma_{yz} &= (\theta_x + w,_y) \\ &\quad + P_1(\xi)w_{1,y} + [P_0/5+P_2(\xi)]w_{2,y} \\ \varepsilon_z &= (w_1 + 3P_1(\xi)w_2)/h \end{aligned}\right] \quad (4)$$

FIELD-CONSISTENT DISPLACEMENT GRADIENTS

A. Shear strains

$$\left.\begin{aligned} u^*_{x,z} &= \sum_{k=0}^{2} a_k(x,y)\, P_k(\xi) \\ u^*_{y,z} &= \sum_{k=0}^{2} b_k(x,y)\, P_k(\xi) \end{aligned}\right] \quad (5)$$

SUBJECT TO:

(a) Shear Traction-Free on S^+ & S^-

$$\left.\begin{aligned} \gamma_{xz}(\xi=\pm1) &= \{u^*_{x,z} + u_{z,x}\}\big|_{\xi=\pm1} = 0 \\ \gamma_{yz}(\xi=\pm1) &= \{u^*_{y,z} + u_{z,y}\}\big|_{\xi=\pm1} = 0 \end{aligned}\right] \quad (6)$$

(b) Mean Equivalence:

$$\min \int_{-h}^{h} (u^*_{x',z} - u_{x',z})^2 \, dz$$

$$\min \int_{-h}^{h} (u^*_{y',z} - u_{y',z})^2 \, dz \qquad (7)$$

Field-consistent Shear Strains

$$\gamma_{xz} = k^2(P_0 - P_2)(w_{,x} + \theta_y)$$

$$\gamma_{yz} = k^2(P_0 - P_2)(w_{,y} + \theta_x) \qquad (8)$$

$$(k^2 = 5/6)$$

B. <u>Transverse Normal Strain</u>

$$\varepsilon_z = u^*_{z',z} = \sum^{3} \frac{1}{h} c_k(x,y)\, P_k(\xi) \qquad (9)$$

(a) Stress Boundary Condition:

$$\frac{\partial}{\partial z}(\sigma_z)\Big|_{\xi=\pm 1} = 0 \qquad (10)$$

(b) Mean Equivalence:

$$\min \int_{-h}^{h} (u^*_{z',z} - u_{z',z})^2 \, dz \qquad (11)$$

ε_z emerges as:

$$\varepsilon_z = \frac{1}{h} \{w_1 + k_z^2 \, [(6P_1 - P_3)w_2 - \frac{1}{42}(14P_3 + P_1)(C_{31}\theta_{y,x} + C_{32}\theta_{x,y})]\} \qquad (12)$$

$k_z^2 = 42/85, \quad C_{ij}$ = elastic moduli

<u>VIRTUAL WORK</u>

$$\iiint_V (\sigma_x \delta\varepsilon_x + \sigma_y \delta\varepsilon_y + \sigma_z \delta\varepsilon_z + \tau_{xy}\delta\gamma_{xy} + \tau_{xz}\delta\gamma_{xz} + \tau_{yz}\delta\gamma_{yz})\, dxdydz$$

$$- \iint_{S^+} q^+ \delta u_z dxdy - \iint_{S^-} q^- \delta u_z dxdy \qquad (13)$$

$$- \iint_{S_\sigma} (\bar{T}_x \delta u_x + \bar{T}_y \delta u_y + \bar{T}_z \delta u_z)\, dsdz = 0$$

<u>STRESS & MOMENT RESULTANTS</u>

$$q_1 = q^+ - q^-, \quad q_2 = q^+ + q^-$$

$$(N_x, N_y, N_{xy}) = \int_{-h}^{h} (\sigma_x, \sigma_y, \tau_{xy})\, dz$$

$$(M_x, M_y, M_{xy}) = \int_{-h}^{h} (\sigma_x, \sigma_y, \tau_{xy}) z \, dz$$

$$(Q_x, Q_y) = \int_{-h}^{h} k^2 (P_0 - P_2)(\tau_{xz}, \tau_{yz}) dz$$

$$N_z = \int_{-h}^{h} \sigma_z \, dz, \quad M_z = \int_{-h}^{h} \sigma_z k_z^2 \, (6P_1 - P_3) dz$$

$$(\bar{N}_{xn}, \bar{N}_{yn}) = \int_{-h}^{h} (\bar{T}_x, \bar{T}_y)\, dz \qquad (14)$$

$$(\bar{M}_{xn}, \bar{M}_{yn}) = \int_{-h}^{h} (\bar{T}_x, \bar{T}_y) z \, dz$$

$$\bar{Q}_{zn} = \int_{-h} \bar{T}_z \, dz$$

$$N_{xn} = N_x \ell + N_{xy} m, \quad N_{yn} = N_{xy} \ell + N_y m$$

$$M_{xn} = M_x \ell + M_{xy} m, \quad M_{yn} = M_{xy} \ell + M_y m$$

$$Q_{zn} = Q_x \ell + Q_y m$$

$$\ell = \cos(x,n), \quad m = \cos(y,n)$$

EQUILIBRIUM EQUATIONS

$$
\begin{aligned}
(\delta u):&\quad N_{x,x} + N_{xy,y} = 0\\
(\delta v):&\quad N_{xy,x} + N_{y,y} = 0\\
(\delta \theta_y):&\quad M_{x,x} + M_{xy,y} - Q_x = 0\\
(\delta \theta_x):&\quad M_{xy,x} + M_{y,y} - Q_y = 0\\
(\delta w):&\quad Q_{x,x} + Q_{y,y} + q_1 = 0\\
(\delta w_1)&\quad -N_z/h + q_2 = 0\\
(\delta w_2):&\quad -M_z/h + q_1/k^2 = 0
\end{aligned}
\qquad (15)
$$

BOUNDARY CONDITIONS

On C_1:

$$
N_{xn} = \bar{N}_{xn},\quad N_{yn} = \bar{N}_{yn},
$$

$$
M_{xn} = \bar{M}_{xn},\quad M_{yn} = \bar{M}_{yn},\quad Q_{zn} = \bar{Q}_{zn} \qquad (16)
$$

On C_2:

$$
u = \bar{u},\quad v = \bar{v},\quad \theta_y = \bar{\theta}_y,\quad \theta_x = \bar{\theta}_x,\quad w = \bar{w} \qquad (17)
$$

HOOKE'S LAW

$$
\begin{aligned}
N_x &= \bar{A}_{11}u_{,x} + \bar{A}_{12}v_{,y} + \bar{A}_{13}\frac{w_1}{h}\\
N_y &= \bar{A}_{12}u_{,x} + \bar{A}_{22}v_{,y} + \bar{A}_{23}\frac{w_1}{h}\\
N_{xy} &= A_{66}(u_{,y} + v_{,x})\\
M_x &= \bar{D}_{11}\theta_{y,x} + \bar{D}_{12}\theta_{x,y} + \bar{D}_{13}\frac{w_2}{h}\\
M_y &= \bar{D}_{12}\theta_{y,x} + \bar{D}_{22}\theta_{x,y} + \bar{D}_{23}\frac{w_2}{h}\\
M_{xy} &= D_{66}(\theta_{x,x} + \theta_{y,y})\\
Q_x &= G_{55}(w_{,x} + \theta_y)\\
Q_y &= G_{44}(w_{,y} + \theta_x)\\
N_z &= \bar{A}_{13}u_{,x} + \bar{A}_{23}v_{,y} + \bar{A}_{33}\frac{w_1}{h}\\
M_z &= \bar{D}_{13}\theta_{y,x} + \bar{D}_{23}\theta_{x,y} + \bar{D}_{33}\frac{w_2}{h}
\end{aligned}
\qquad (18)
$$

where A_{ij}, D_{ij}, $\bar{A}_{ij}$, $\bar{D}_{ij}$ and G_{ij} are elastic plate rigidities

SOLUTIONS OF 'AUXILIARY' DISPLACEMENTS

$$
\begin{aligned}
\frac{w_1}{h} &= \frac{1}{\bar{A}_{33}}\left[q_2 h - \bar{A}_{13}u_{,x} - \bar{A}_{23}v_{,y} \right]\\
\frac{w_2}{h} &= \frac{1}{\bar{D}_{33}}\left[q_1 \frac{h}{k^2} - \bar{D}_{13}\theta_{y,x} - \bar{D}_{23}\theta_{x,y} \right]
\end{aligned}
\qquad (19)
$$

EQUILIBRIUM EQUATIONS

$$
\begin{aligned}
&A_{11}u_{,xx} + A_{12}v_{,xy} + A_{66}(u_{,yy} + v_{,xy}) + \frac{C_{13}}{C_{33}}(hq_2)_{,x} = 0\\
&A_{12}u_{,xy} + A_{22}v_{,yy} + A_{66}(u_{,xy} + v_{,xx}) + \frac{C_{23}}{C_{33}}(hq_2)_{,y} = 0
\end{aligned}
\qquad (20)
$$

$$
\begin{aligned}
&D_{11}\theta_{y,xx} + D_{12}\theta_{x,xy} + D_{66}(\theta_{x,xy} + \theta_{y,yy}) - G_{55}(w_{,x} + \theta_y) + \frac{2C_{13}}{5C_{33}}(q_1 h^2)_{,x} = 0\\
&D_{12}\theta_{y,xy} + D_{22}\theta_{x,yy} + D_{66}(\theta_{x,xx} + \theta_{y,xy}) - G_{44}(w_{,y} + \theta_x) + \frac{2C_{23}}{5C_{33}}(q_1 h^2)_{,y} = 0\\
&G_{55}(w_{,xx} + \theta_{y,x}) + G_{44}(w_{,yy} + \theta_{x,y}) + q_1 = 0
\end{aligned}
\qquad (21)
$$

TRANSVERSE STRESSES

$$
\left.
\begin{aligned}
\sigma_z &= \frac{1}{4}\left[q_1\xi(3-\xi^2)+2q_2\right] \\
\tau_{xz} &= \frac{3k^2}{2}C_{44}(1-\xi^2)(w_{,x}+\theta_y) = \frac{3}{4h}Q_x(1-\xi^2) \\
\tau_{yz} &= \frac{3k^2}{2}C_{55}(1-\xi^2)(w_{,y}+\theta_x) = \frac{3}{4h}Q_y(1-\xi^2)
\end{aligned}
\right] \quad (22)
$$

FINITE ELEMENT APPLICABILITY

- SIMPLE & EFFICIENT ELEMENTS DUE TO:

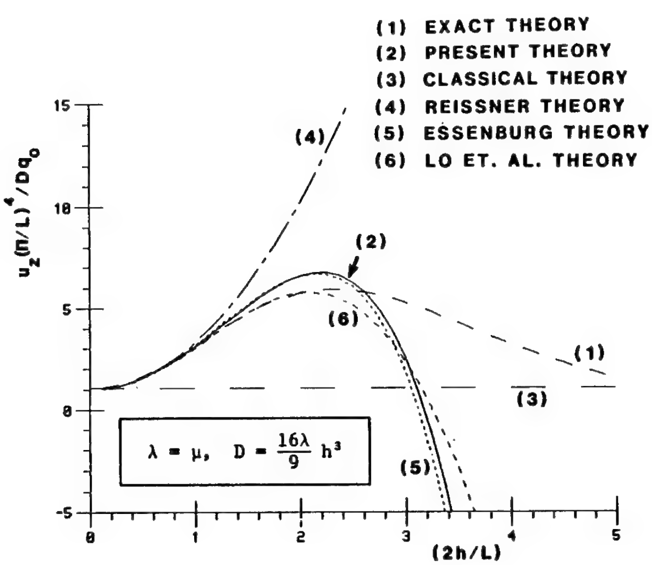
  C^0- continuity of u, v, w, θ_x, θ_y
  C^{-1}-continuity of w_1 and w_2

- FEM METHODOLOGY OF REISSNER-MINDLIN PLATES APPLIES [15-17]

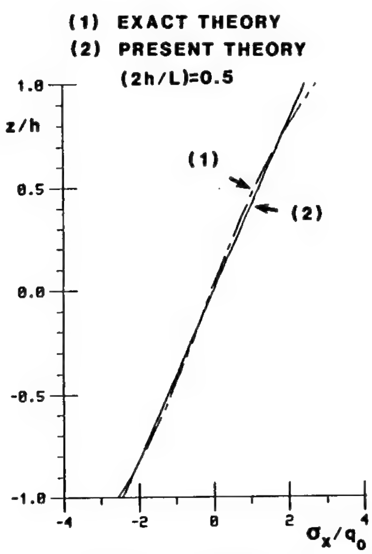
ASSESSMENT OF THEORY

- INFINITE PLATE UNDER SINUSOIDAL PRESSURE

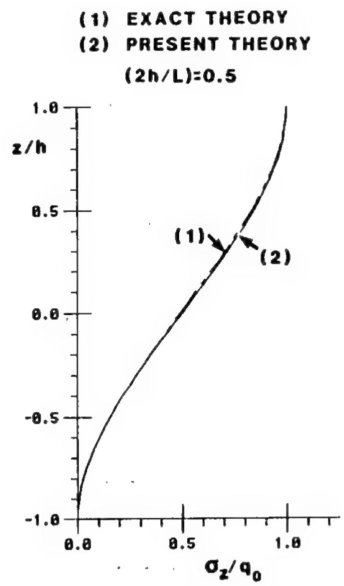
  $q^+ = q_0 \sin(\pi x/L)$ on S^+ ($q^- = 0$ on S^-)



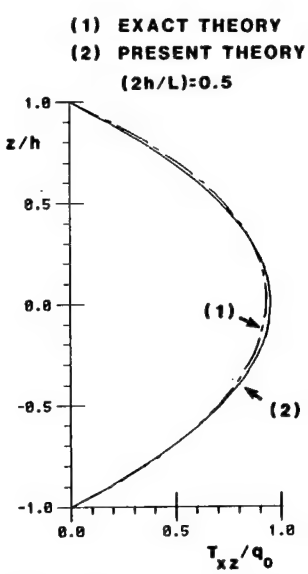
MAXIMUM MIDPLANE TRANSVERSE DISPLACEMENT



MAXIMUM NORMAL STRESS ACROSS THICKNESS



MAXIMUM TRANSVERSE STRESS ACROSS THICKNESS



MAXIMUM SHEAR STRESS ACROSS THICKNESS

CONCLUDING SUMMARY

- 10TH-ORDER DISPLACEMENT THEORY DECOUPLED INTO 4TH ORDER MEMBRANE & 6TH ORDER BENDING EQUIL. EQUATIONS
- FIELD CONSISTENCY IN TRANSVERSE SHEAR
- PARABOLIC TRANSVERSE SHEAR & CUBIC TRANSVERSE NORMAL STRESSES SATISFYING APPROPRIATE BOUNDARY CONDITIONS
- IDEAL FEM APPLICABILITY

IMPLICATIONS OF ADVANCED 3-D ANALYTICAL METHODS

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ABSTRACT

Three-dimensional behavior of thick sectioned laminated composite structures is generally characterized by induced strains and stresses acting in the out-of-plane directions of the plies. This response is strongly influenced by anisotropic properties of the overall laminate, which arises from interactions between plies of varying fiber orientation and stiffness. Further complications arise when the load paths are multi-directional, resulting in coupled direct and shear components along the principal material axes.

Analytical evaluation of composites involving three-dimensional structural response, requires models that are sensitive to the interaction between plies within the laminate, and must provide for capability to extract the response of individual plies to determine the critical states of the structure. While finite element codes providing capability to represent anisotropic materials are available, the detail required to develop models that have sufficient resolution to generate results on a ply-by-ply basis is generally prohibitive due to the modeling complexity involved, and the resulting model size. To overcome this difficulty, a generalized three-dimensional laminated material pre- and post-processor is used, which provides capability for representing the as-fabricated ply-by-ply composite in sufficient detail such that accurate response of the structure can be predicted while maintaining reasonably sized models. The processor utilizes the mathematical algorithm defining the deformation field of individual elements in the model to generate consistent 21-independent component material stiffness tensors for each element, reflecting the local biasing and stiffness gradients arising from ply content within the element. Post-processing of results is accomplished by extracting the predicted deformation field local to each element and computing the resulting strains along the plies contained within the element's volume.

An example 3-D analysis of an all-composite pinjoint is presented to illustrate capabilities of the processor. Discussion of ensuing strain states arising in typical three-dimensional composite response is undertaken to describe critical failure modes and implications to design and fabrication methods. Some general observations are then made regarding failure mechanisms in composites and the methods used for assessing various predicted critical states of strain and stress against characteristic failure modes.

ACKNOWLEDGEMENTS

1. Portions of this work were sponsored by U.S. Army Ballistic Research Laboratory, Aberdeen Proving Ground, MD 21005.

2. Work performed on the all-composite pinjoints was conducted in association with Hercules Incorporated, Allegany Ballistics Laboratory, Rocket Center, WV 26726.

IMPLICATIONS OF ADVANCED 3-D ANALYTICAL METHODS

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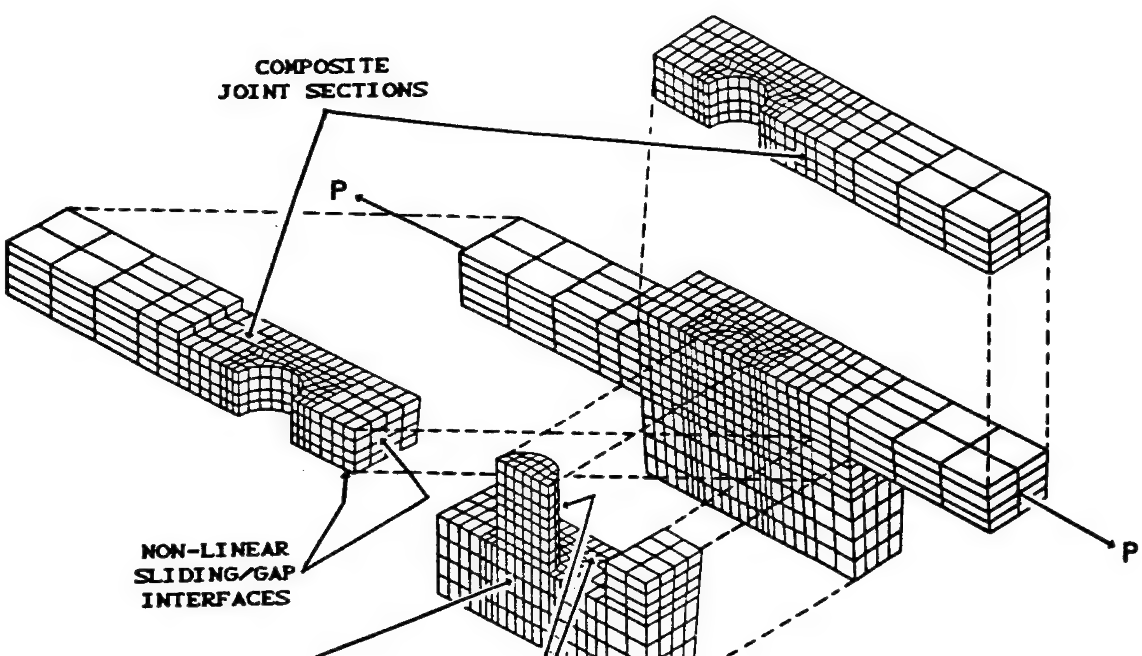
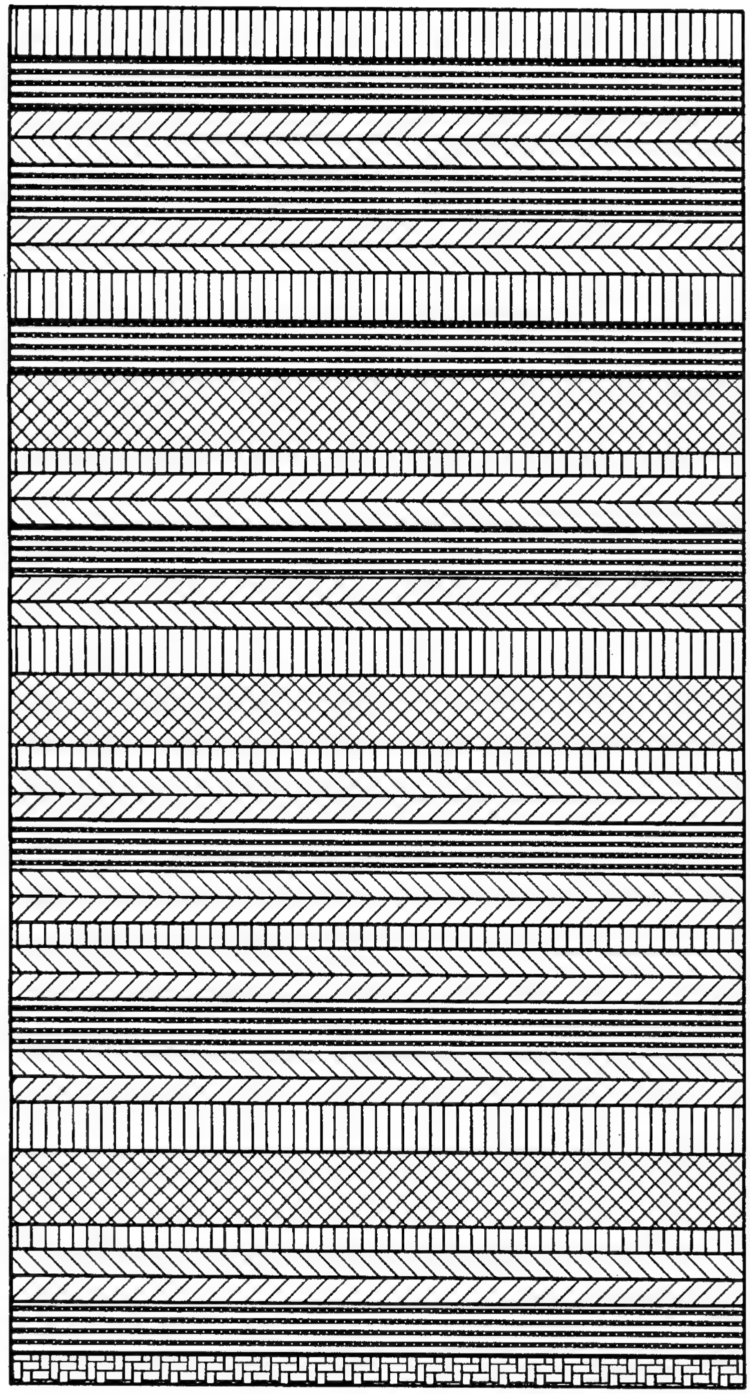
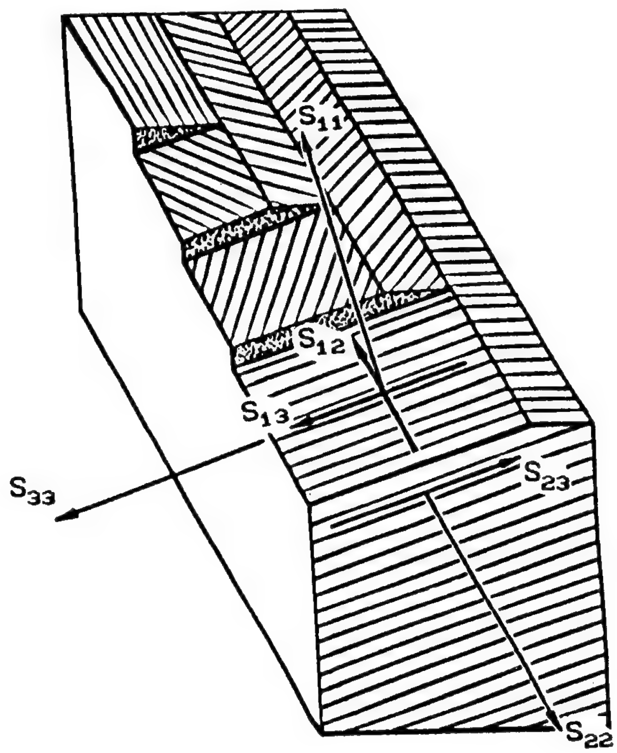
3-D MODELING OF AN ALL-COMPOSITE PINJOINT



ALL-COMPOSITE PINJOINT LAYUP



GENERALIZED 3-D LAMINATED MATERIAL PRE- AND POST-PROCESSOR



PREPROCESSOR: DEVELOPS ACCURATE REPRESENTATION OF AS-FABRICATED PLY-BY-PLY COMPOSITE

- FABRICATION LAYUP SCHEDULE INPUT BY USER

  PLY LAYUP SEQUENCE

  INDIVIDUAL PLY DATA – DIMENSIONS, MATERIAL, THICKNESS, FIBER ORIENTATION, PLACEMENT

- PLY CONTENT OF EACH ELEMENT IN MODEL DETERMINED

- STRAIN ENERGY CONTRIBUTION OF EACH PLY IN ELEMENT COMPUTED

  USES ELEMENT DEFORMATION SHAPE FUNCTION

- EQUIVALENT HOMOGENEOUS 21 INDEPENDENT COMPONENT STIFFNESS TENSOR COMPUTED AT EACH ELEMENT INTEGRATION POINT

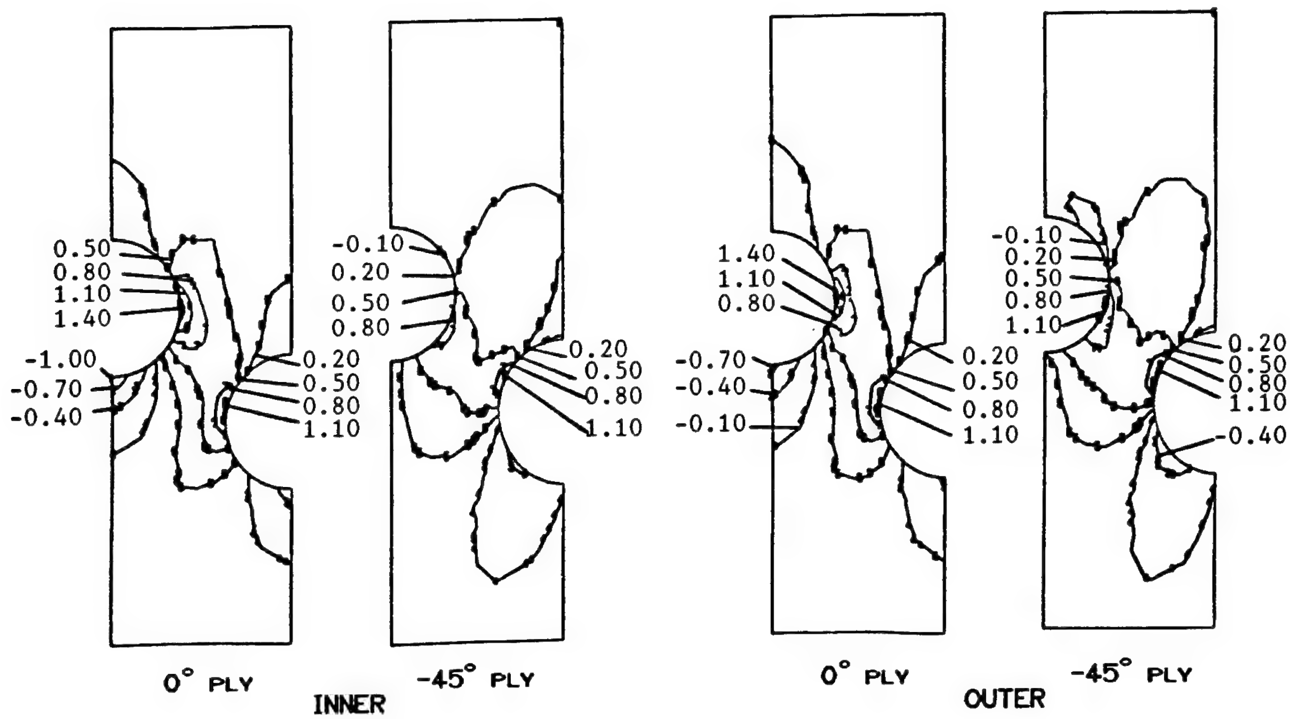
  LOCAL STIFFNESS BIASING PRESERVED

POSTPROCESSOR: EXTRACTS PLY-BY-PLY STRESS AND STRAIN RESPONSE USING PREDICTED DEFORMATION FIELD OF ELEMENT

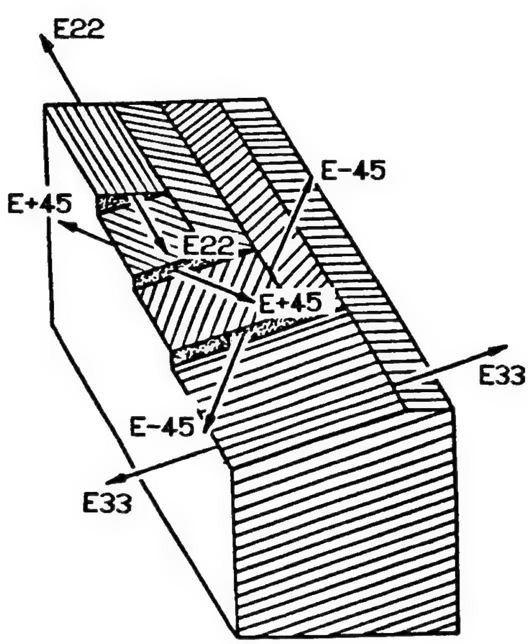
- ACCURATE EVALUATION OF STATE VARIABLES AT CRITICAL REGIONS

- PROVIDES CAPABILITY TO DETERMINE INTERACTION OF PLIES

PREDICTED STRAIN FIELDS IN CRITICAL PLIES OF PINJOINT



EXAMPLES OF PREDICTED CRITICAL STRAIN STATES



TENSION DOMINATED
(1.5% ULTIMATE ALONG FIBER)

	E22	E33	E+45	E-45
1.	1.04%	-0.88%	0.24%	-0.08%

COMPRESSION DOMINATED
(-0.55% TO -1.10% ULTIMATE ALONG FIBER)

	E22	E33	E+45	E-45
1. A	-0.82%	-0.03%	0.27%	-1.12%
B	-0.39%	-0.34%	0.36%	-1.10%
2.	-1.06%	0.34%	-0.40%	-0.32%

FAILURE MECHANISMS

CRITICAL STATE PARAMETERS:

FIBER STRAINS

LAMINATE STRESSES – NORMAL, INTERLAMINAR

LOAD PATHS – PRINCIPAL, SECONDARY

INTERFIBER LOAD TRANSFER MECHANISM

FAILURE STATES IN MULTI-ANGLE PLY LAMINATES:

▸ TENSILE FIBER STRAIN ALONG PRIMARY FIBERS TO PRINCIPAL TENSILE LOAD PATH	▸ MAXIMUM FIBER STRAIN CRITERION "RAPID" FAILURE PROPAGATION
▸ COMPRESSIVE FIBER STRAIN ALONG PRIMARY FIBERS TO PRINCIPAL COMPRESSIVE LOAD PATH	▸ FIBER BUCKLING (SENSITIVE TO LATERAL STRESSES) "SLOW" PROGRESSIVE FAILURE TOWARD UNSTABLE LOAD REDISTRIBUTION TO OTHER PLIES
▸ COMPRESSIVE FIBER STRAIN ALONG SECONDARY FIBERS TO PRINCIPAL TENSILE OR COMPRESSIVE LOAD PATH	▸ LOCAL FIBER BUCKLING WITH LOAD REDISTRIBUTION. RESIN CRAZING. REDUCTION IN INTERLAMINAR SHEAR STRENGTH.
▸ INTERLAMINAR SHEAR ALIGNED WITH PRINCIPAL LOAD PATH	▸ SENSITIVE TO NORMAL STRESS.

CONCLUSIONS AND RECOMMENDATIONS

- ANISOTROPIC FAILURE MECHANISMS IN MULTI-ANGLE PLY COMPOSITES REQUIRE DIFFERING FAILURE CONSTITUTIVE MODELS DEPENDING ON MODE OF FAILURE.

- A PROGRESSIVE FAILURE MODEL IS NEEDED, ACCOUNTING FOR:

  INTER-PLY LOAD REDISTRIBUTION
  PRINCIPAL LOAD PATHS IN COMPONENT
  REDUNDANCY IN LOAD CARRYING MECHANISMS

- MAXIMUM FIBER STRAIN CRITERION IS APPROPRIATE FOR TENSION DOMINATED STRAIN STATES

- COMPRESSION DOMINATED STRAIN STATES REQUIRE AN IN-DEPTH EVALUATION OF THE FIBER BUCKLING FAILURE PATH USING A STABILITY CRITERION

STUDIES OF MECHANICAL TEST METHODS FOR COMPOSITES*

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ABSTRACT

Anisotropy and nonhomogeneity of fiber reinforced composite laminates introduce considerable complexity in testing. Problems arise due to complex damage mechanisms and the difficulty in attaining uniform stress states in specimens. In Phase I of this work (ref. 1) attention was focussed on analytical studies of various specimens for characterizing in-plane shear response. A combined analytical/experimental program is being conducted in Phase II to evaluate specimens for characterizing properties under tension, compression, in-plane, and interlaminar shear as well as combined stress states.

Rail Shear, Iosipescu Shear, Torsion, and ±45° Tension specimens were considered for in-plane shear tests. The single rail shear (ASTM D4255 guideline) specimen with tapered rails is found to be better than that with rails of constant thickness. However, it appears that a parallelogram shaped test section may yield further improvement over rectangular shaped ones. Effect of material nonlinearity does not alter the stress distribution very much for unidirectional materials. Effects of loading fixture and matrix nonlinearity on response of Iosipescu specimens were also found to be of little importance. However, a larger specimen (3" x 0.75") yields better stress distribution in some cases. Rounded notches with 90° and 120° notch angles have been selected for the experimental study. Tests on Rail Shear and Iosipescu specimens (two geometries each) for various laminates as well as torsion and ±45° tension tests for unidirectional material properties are being conducted at the University of Wyoming. Interlaminar shear tests with Iosipescu and Short Beam Shear specimens are also planned.

Compression tests will be conducted on IITRI, Celanese, End Loaded Side Supported and Sandwich Beam specimens. Analyses were performed for determining buckling loads of some of these specimens with due consideration to material nonlinearity and initial imperfections. The results were correlated with test data reported in literature and it appears that specimen, ply or microbuckling are strongly influenced by these factors. However, results from different specimens are found to be close to one another although failure loads depend on the slenderness ratios. Tests for two slenderness ratios (10 and 20) are suggested to determine material properties needed for design purposes.

Combined stress effects will be studied by using tension tests on off-axis and $\pm\theta$ laminates. A phenomenological damage model with or without plastic response of laminae will be utilized for data analysis. Tension tests on ASTM D3039, Linear Taper, and Streamline specimens will be performed for 0° and 0/90 laminates and their responses will also be examined.

REFERENCES

1. V. Ramnath, and S. N. Chatterjee, "Composite Specimen Design Analysis," MSC TFR 1701/1703, Final Report under Contract DAAG46-85-C-0058 for Army Materials Technology Laboratory, March, 1986.

*Work Supported by Army Materials Technology Laboratory, Watertown, MA 02172 Under Contract DAAL04-87-C-0064.

STUDIES OF MECHANICAL TEST METHODS FOR COMPOSITES*

MATERIALS SCIENCES CORPORATION
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*Work Performed Under AMTL Contract DAAL04-87-C-0064

Materials Sciences Corporation

OBJECTIVES

- Combined Analytical/Experimental Program
  - To Evaluate Suitability of Various Test Methods
  - To Recommend Improved Test Methods and Optimal Specimens
  - To Suggest Simple Data Interpretation Procedures
- Tests Considered
  - In-Plane Shear
  - Compression
  - Tension
  - Interlaminar Shear
  - Combined Stress
  - Interlaminar Fracture

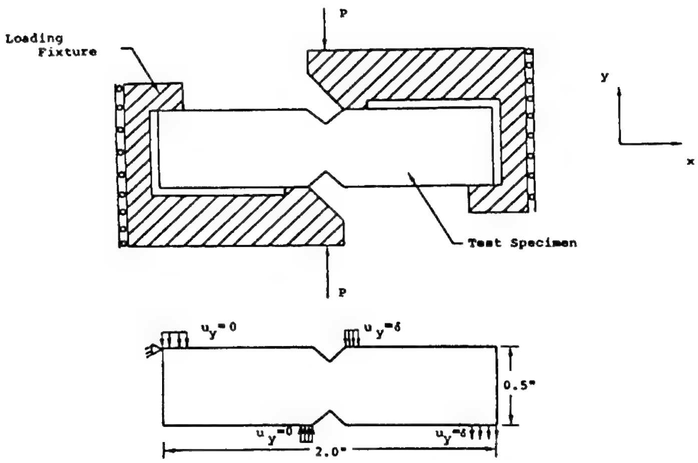
IN-PLANE SHEAR TEST METHODS CONSIDERED

- Iosipescu
  - Lamina and Laminates
  - Effects of Notch Angle and Fillet
  - Effects of Loading Fixture on Stress Distribution
  - Effects of Matrix Nonlinearity
- Single Rail Shear
  - Lamina and Laminates
  - Standard, With and Without Tapered Rails
  - Modified, With Tapered Rails
  - Effects of Matrix Nonlinearity
- Torsion
  - Lamina
  - Rectangular Specimens
  - Data Reduction for Nonlinear Response
- $(\pm 45)_{ns}$ Tension
  - Lamina
  - Effects of Lay-Up Sequence on Criticality of Edge Delaminations and Transverse Cracking

IOSIPESCU ANALYSES

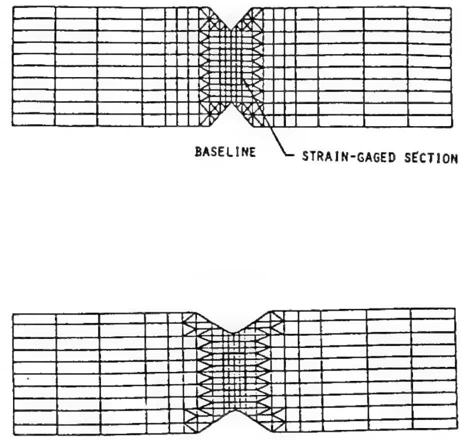
- Elastic Analyses of Specimen Without the Fixture Indicated Notch Angle of 120° With Fillet Produced Improved Stress Distributions
- Effects of Loading Fixture on Modified Geometry Specimen
  - Not Allowing Slipping at Specimen/Fixture Interface
  - Allowing Slipping at Specimen/Fixture Interface
- Effects of Matrix Nonlinearity
  - Incremental Elastoplastic Analyses of 0° Specimen Without Fixture
  - Matrix Modeled Using Bilinear Stress-Strain Curve
- Specimen Sizes 2" x 0.5" and 3" x 0.75"
  - 3" x 0.75" With 120° and 90° Notch With Fillet Will be Used in Tests

THE IOSIPESCU SHEAR TEST METHOD



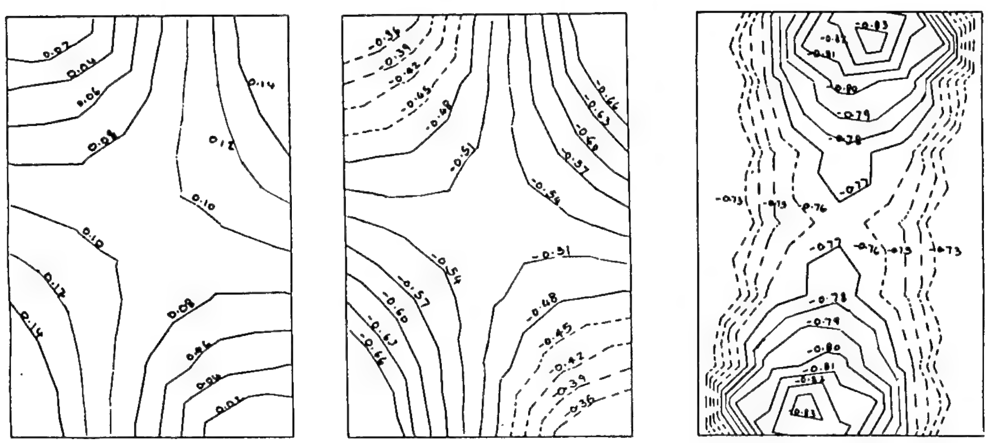
Typical Specimen Geometry and Boundary Conditions

FINITE ELEMENT MODELS OF ISOIPESCU SPECIMEN



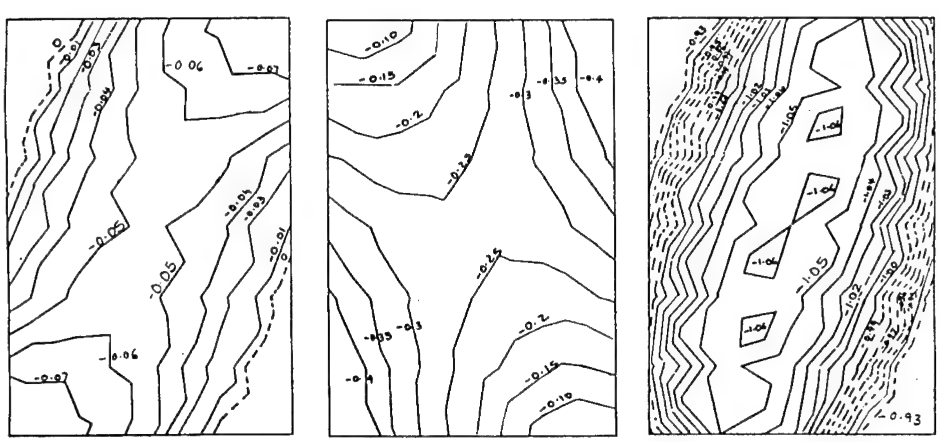
MODIFIED GEOMETRY

STRESS CONTOURS IN STRAIN-GAGED SECTION FOR BASELINE GEOMETRY, 0° T300/EP



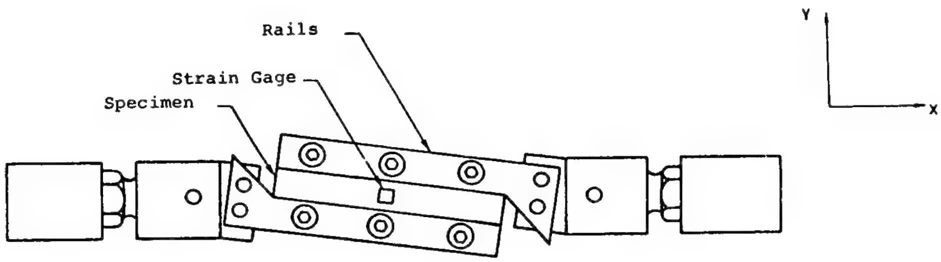
SIGX SIGY TAUXY

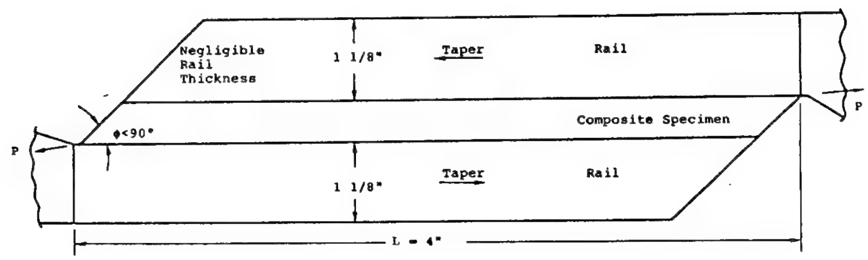
STRESS CONTOURS IN STRAIN-GAGED SECTION FOR MODIFIED GEOMETRY, 0/±45/90 T300/EP



SIGX SIGY TAUXY

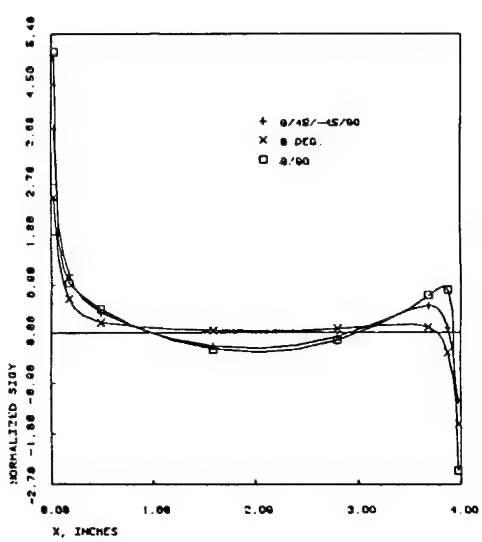
SINGLE RAIL SHEAR TEST - STANDARD AND MODIFIED GEOMETRIES

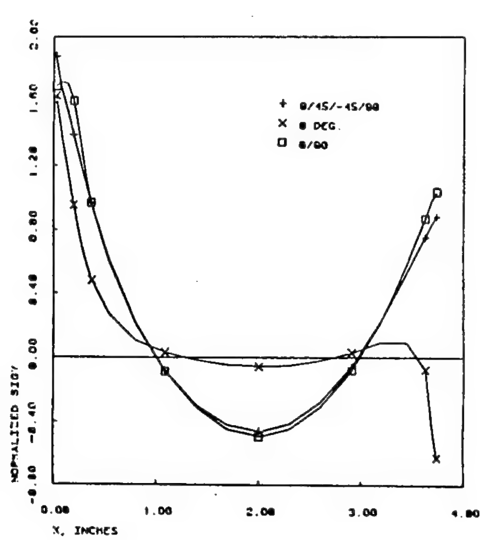




- ELASTIC ANALYSES DONE USING MODIFIED GEOMETRY AND TAPERED RAILS
  - TWO DIFFERENT LENGTHS: 4 IN. AND 6 IN.
  - TWO DIFFERENT ANGLES: 45° AND 60°
- RECOMMEND ϕ = 45° AND L = 4 IN.

STRESS DISTRIBUTION ALONG LENGTH NEAR RAILS - BASELINE AND MODIFIED (Φ=60°) RAIL SHEAR SPECIMENS

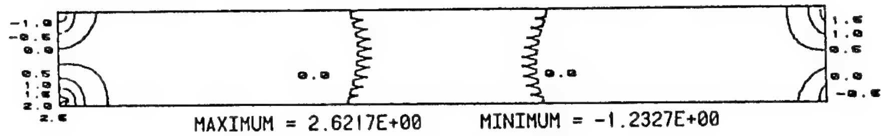




STRESS CONTOUR PLOTS FOR 0° RAIL SHEAR TEST SPECIMEN, TAPERED RAILS
(T300/EPOXY)

MAXIMUM = 9.8344E-01 MINIMUM = -1.6785E-01

STRAIN GAGE SECTION SIG-XX = 5.8707E-01



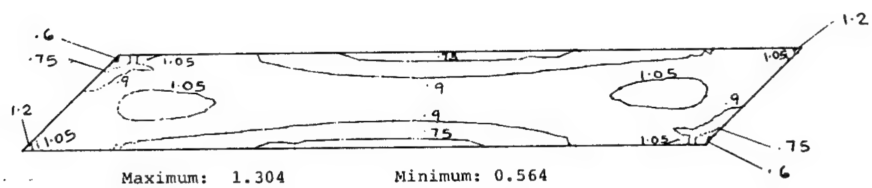
MAXIMUM = 2.6217E+00 MINIMUM = -1.2327E+00

SIG-YY = -2.6647E-02

MAXIMUM = 1.1093E+00 MINIMUM = 1.5379E-01

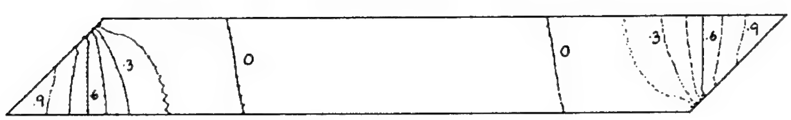
TAU-XY = 1.0430E+00

STRESS CONTOUR PLOTS FOR 0° RAIL SHEAR TEST SPECIMEN, MODIFIED GEOMETRY (ϕ = 45°)



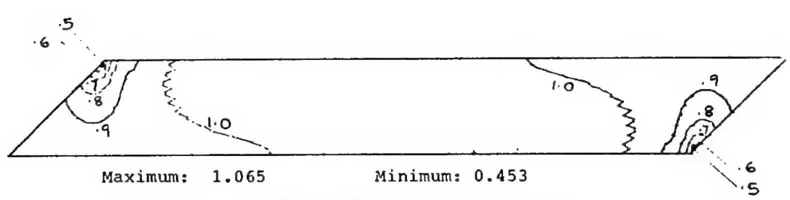
Maximum: 1.304 Minimum: 0.564

Gage-Section: SIG-XX = 0.709



Maximum: 0.987 Minimum: -0.082

Gage-Section: SIG-YY = -0.077

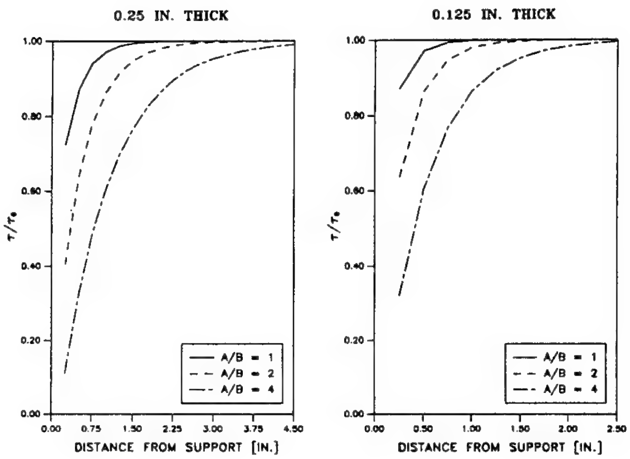


Maximum: 1.065 Minimum: 0.453

Gage-Section: TAU-XY = 1.051

TORSION OF RECTANGULAR BAR -

STRESS DECAY



COMPRESSION TESTING

- Bending of Sandwich Beams is a Reliable Test Method But is Expensive to Fabricate and Test

- Evaluation of Other More Commonly Used Test Methods

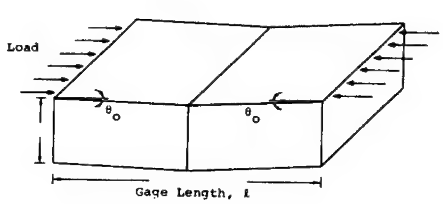
  (IITRI, Celanese, ASTM-D695, RAE)

  - Do These Tests Yield Reasonable Data?
  - What are the Typical Failure Mechanisms?
  - Guidelines for Reliable Data

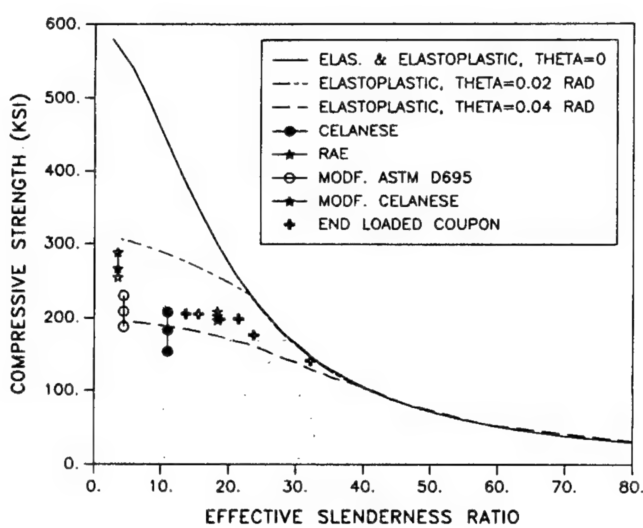
ANALYTICAL STUDIES OF BUCKLING

- Approach Includes
  - Shear Deformation
  - Inelastic Effects
  - Effects of Imperfections

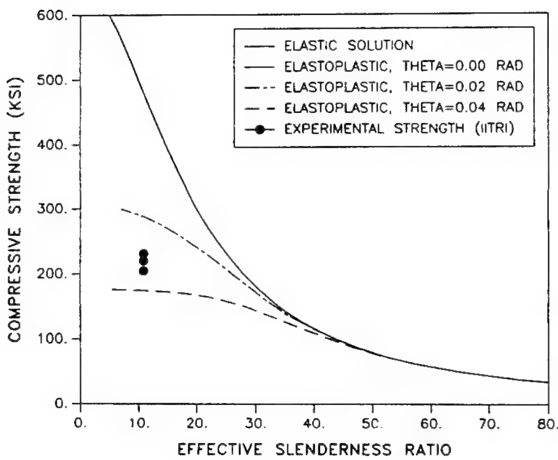
- "EPLAM" (Elastoplastic Laminate) Analysis
  - Nonlinear Layer Stress-Strain Curve Used as Input
  - In-Plane Compression and Transverse Shear Applied (In the Case of Assumed Imperfection)
  - Instantaneous Laminate Bending and Shear Stiffnesses Used in Computing Critical Buckling Stresses



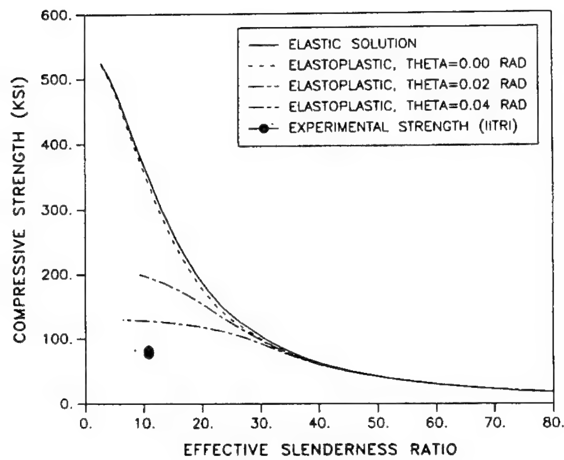
COMPRESSION TEST DATA CORRELATIONS FOR 16 PLY 0° AS-4/914



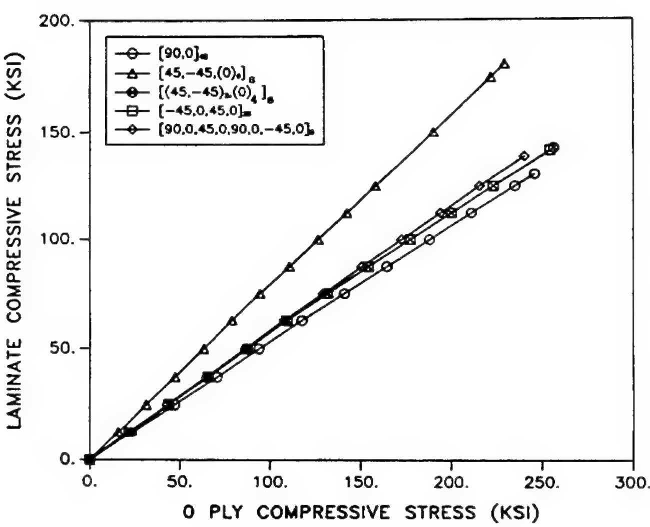
COMPRESSION TEST DATA CORRELATIONS FOR 16 PLY 0° T300/5208



COMPRESSION TEST DATA CORRELATIONS FOR $(0/45/-45/90)_{2S}$ T300/5208

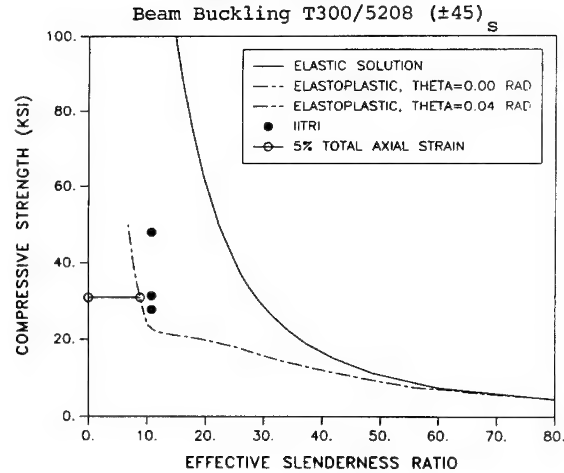


0° PLY STRESSES AT FAILURE FOR VARIOUS AS-4/S14 LAMINATES IN COMPRESSION

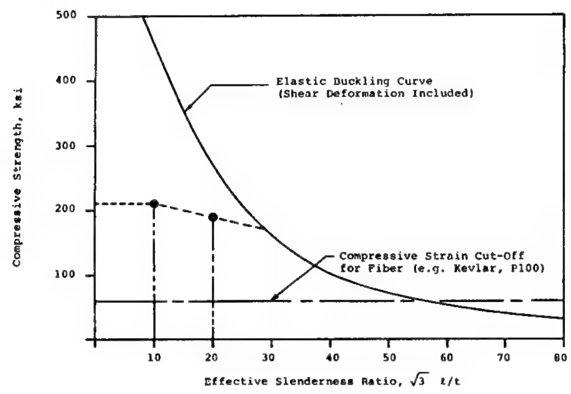


COMPRESSIVE TEST DATA CORRELATIONS

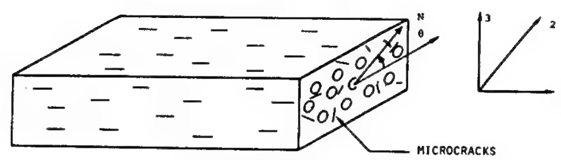
Beam Buckling T300/5208 $(\pm 45)_s$



SUGGESTED PROCEDURE FOR OBTAINING INELASTIC BUCKLING CURVES FROM COMPRESSION TEST DATA



COMBINED STRESS - DAMAGE, PHENOMENOLOGICAL MODEL



ALL CRACK PLANES PARALLEL TO x_1 AXIS

ORIENTATION = θ $\quad 0 < \theta < 2\pi$

$N = [0, \cos\theta, \sin\theta]$ - CRACK ORIENTATION VECTOR

$\omega(N)$ = SOME VOLUME AVERAGE OF AREA OF CRACKS WITH NORMAL N

STRESS FORMULATION - NO PLASTICITY

$$\underline{\varepsilon} = \underline{\varepsilon}(\underline{\sigma}, (\omega^\alpha, N^\alpha), \ \alpha = 1, 2, \ldots n)$$

$$W_C = W_C(\underline{\sigma}, (\omega^\alpha, N^\alpha))$$

$$\varepsilon_i = \frac{\partial W_C}{\partial \sigma_i}; \quad R^\alpha = \frac{\partial W^C}{\partial \omega^\alpha}, \quad \sum_{\alpha=1}^{n} R^\alpha \partial \omega^\alpha > 0$$

$$W_C = \frac{1}{2}\left[S_{ij}\,\sigma_i\,\sigma_j + \sum_{\alpha=1}^{n} \omega^\alpha (C_1 h_\alpha \sigma_\alpha^2 + C_2 \tau_{t\alpha}^2 + C_3 \tau_{a\alpha}^2 + \Sigma \frac{\omega^{\alpha 2}}{2}(D_1 h_\alpha \sigma_\alpha^2 + D_2 \tau_{t\alpha}^2 + D_3 \tau_{a\alpha}^2)\right]$$

$$h_\alpha = 0 \qquad \sigma_\alpha < 0$$

$$= 1 \qquad \sigma_\alpha > 0$$

$$C_1 = \frac{\pi}{G_a}, \; C_2 = \frac{\pi}{2G_t}; \; C_3 = \frac{\pi}{G_a}$$

For Small ω, $D_i = 0 \quad i = 1, 2, 3$

SUPERPOSE DAMAGE GROWTH AND PLASTICITY

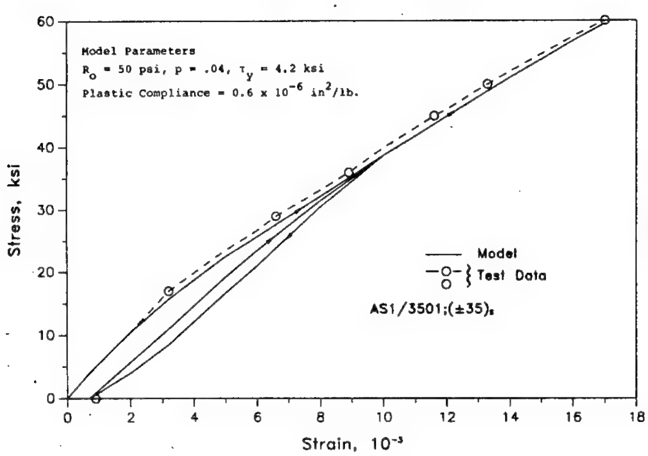
$$\omega^\alpha = (R^\alpha / R_0)^p - 1; \text{ Power Law}$$

$$\varepsilon_i = \varepsilon_i^e + \varepsilon_i^p$$

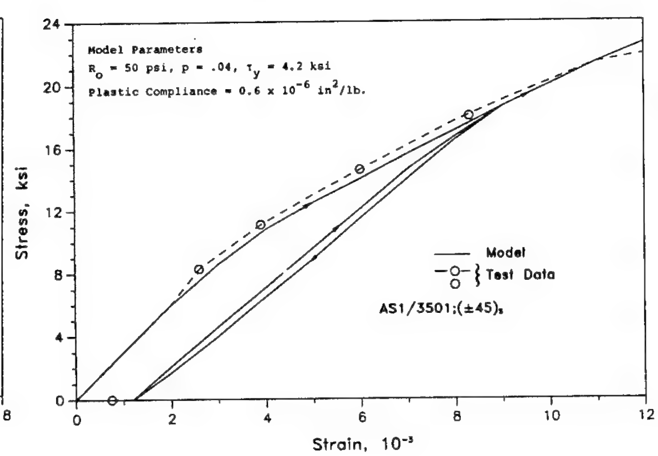
$\dot{\varepsilon}_i^p$ Obtained From Anisotropic Yield Criterion And Associated Flow Rule With Linear Kinematic Hardening

Only One Crack Plane and Interaction of Transverse Extensional And Axial Shear Stresses Are Considered in Data Correlation.

STRESS STRAIN RESPONSE of $(\pm 35)_s$



STRESS STRAIN RESPONSE of $(\pm 45)_s$



CONCLUSIONS AND RECOMMENDATIONS

- Lamina Shear Tests
  - Analyses of (±45) Tension Test (Results Not Shown) Indicate $((\pm 45)_n)_s$ With Dispersed Thin Layers and 1" Width Appears Adequate
  - Effects of Combined Stresses and Nonlinearity Need Investigation
- Lamina and Laminate Shear Tests
  - Iosipescu Specimens With 120° Notch Angle, 20% Notch Depth and 0.05" Root Radius Are Suitable for a Wide Class of Materials. Loading fixture and nonlinearity do not significantly affect gage section stress distributions.
  - Modified Rail Shear Specimens 4" Long With Corner Angles of 45° Are Attractive Alternatives

CONCLUSIONS AND RECOMMENDATIONS (CONT'D)

- Lamina and Laminate Compression Tests
  - Analytical Studies of Buckling Considering Shear Deformation and Imperfections Indicate Ply Buckling or Microbuckling are the Most Probable Causes of Failure in T300/EP and AS-4/EP Materials
  - Imperfections (Fiber Waviness, Load Misalignment and Material Imperfections) Have Substantial Effect on Buckling of Composites
  - Recommend Testing at Two Effective Slenderness Ratios (10 and 20) and Checking for Buckling Failures by Placing Strain Gages at Top and Bottom Surfaces of the Specimen
- Combined Stresses
  - For Small Amount of Damage And Plasticity Phenomenological Approach Appears Satisfactory
- Test Data From the Experimental Program are Awaited

POSTBUCKLING OF ECCENTRIC OPEN-SECTION STIFFENED COMPOSITE PANELS

By

Manuel Stein
NASA Langley Research Center
Hampton, VA 23665-5225

Abstract

A numerical study of the postbuckling behavior of open-section stiffened composite compression panels is presented with emphasis on the effects of an anisotropic attached flange on results, the strain distribution near collapse, and the change of buckle pattern during postbuckling response. Results are obtained for a blade stiffened panel with orthotropic or anisotropic attached flanges from a new version of STAGS, a general, branched shell, nonlinear computer program. Comparisons between these results indicate the effects of anisotropic flanges on the results. Elastic strain distributions are obtained from STAGS and indications are given about the collapse loads and modes of a particular panel. Change of buckle pattern is studied using STAGS and a special purpose computer program for long plates. Comparisons are made with experimental results.

The panel with an anisotropic attached flange has a 10 percent lower buckling load and a lower stiffness at twice the buckling load than the corresponding orthotropic panel. Also, at twice the buckling load, the strains due to twisting are about 30 percent higher for the panel with the anisotropic flange. Numerical buckling results agree with experiment for the analysis of a brittle blade-stiffened laminated panel. At the failure load of the specimen, analysis gives strain measurements of the kind that agree with the experimental mode of failure, and the strains are large and increasing for load near failure. Numerical results are compared with experimental results for an aluminum panel which have changes in buckle pattern. Good agreement was obtained between theory and experiment in the elastic postbuckling range. Thus, this paper has demonstrated that the designer may obtain valuable information on the effects of anisotropic attached flange layup on mode of collapse and imminence of collapse of a stiffened panel, and may determine effects of change in buckle pattern for open section stiffened panels in the postbuckling range from available state-of-the-art computer codes.

POSTBUCKLING OF ECCENTRIC OPEN-SECTION STIFFENED COMPOSITE PANELS

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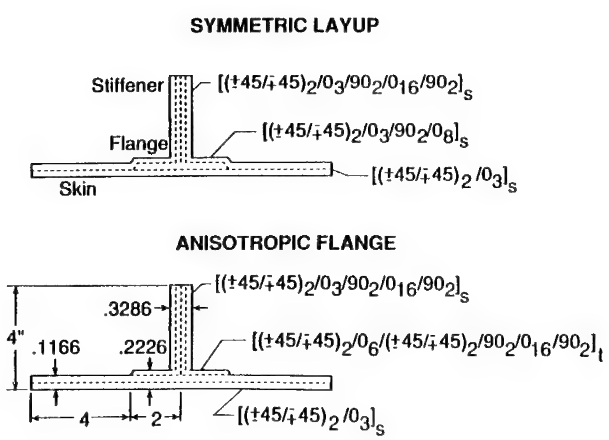
Thirteenth Annual Mechanics of Composites Review

November 2-3, 1988
Bal Harbour, Florida

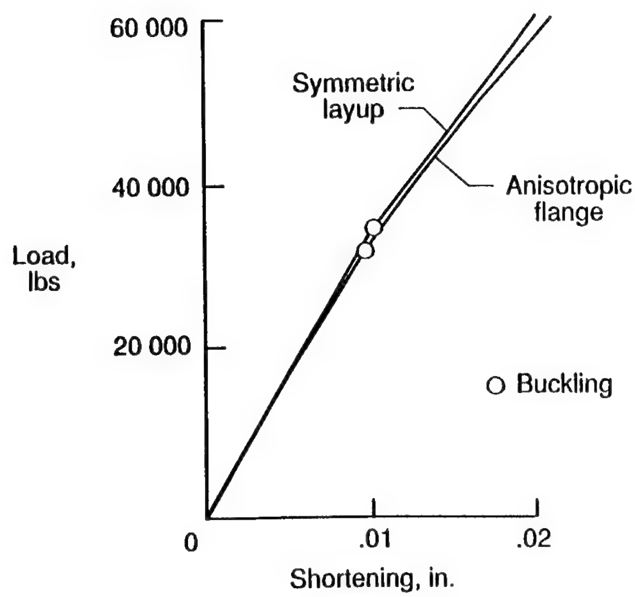
SUBJECTS STUDIED NUMERICALLY

- Effects of using an anisotropic layup in the attached flange in conjunction with balanced symmetric orthotropic layups in the skin and blade stiffener
- Postbuckling strain distribution near collapse
- Change in buckle pattern

BLADE STIFFENED PANEL

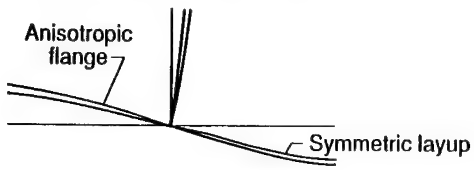


CHARACTERISTIC CURVES

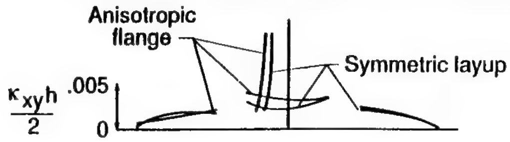


CROSS SECTIONAL DEFORMATIONS

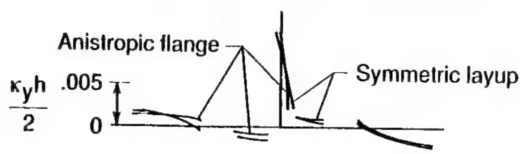
DEFLECTIONS AT MIDLENGTH



TWISTING STRAINS NEAR LOADED BOUNDARY



BENDING STRAINS AT MIDLENGTH

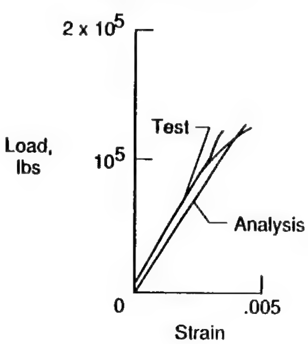


USING ANISOTROPIC INSTEAD OF BALANCED SYMMETRIC LAYUP FOR ATTACHED FLANGE

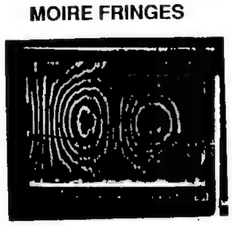
- **At buckling**
  - **10 percent lower buckling load**
- **At twice the buckling load**
  - **Lower postbuckling stiffness**
  - **Strains due to twisting: 30 percent higher**

TEST AND ANALYSIS

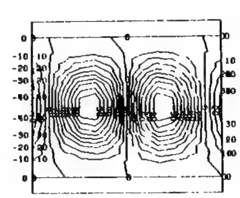
STRAIN AT GAGE



MOIRE FRINGES

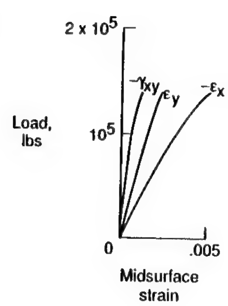


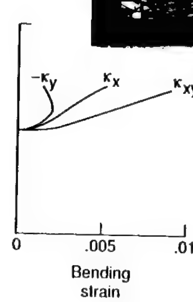
DEFLECTION CONTOURS



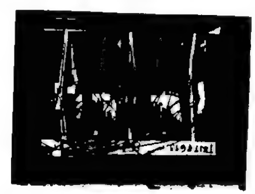
STRAINS NEAR COLLAPSE

MAXIMUM STRAINS
(ON CENTRAL STIFFENER)





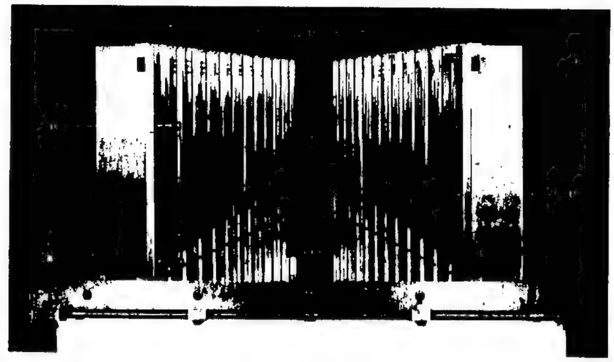
PANEL AFTER COLLAPSE



RESULTS FOR BLADE STIFFENED PANEL OF BRITTLE MATERIAL

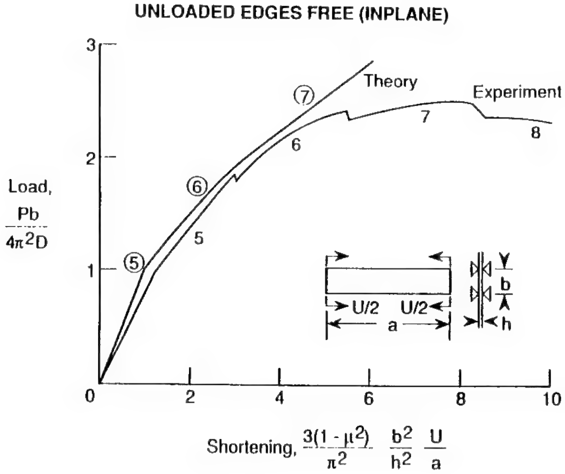
- At buckling
  - Numerical results agree with experimental results
- At collapse load
  - Analysis indicates same collapse mode as experiment
  - Strains are large and increasing with load

BASE PLATES OF MULTI-BAY FIXTURE



CHARACTERISTIC CURVES FOR ALUMINUM PLATE

UNLOADED EDGES FREE (INPLANE)



DESIGNER MAY OBTAIN VALUABLE INFORMATION FROM STATE-OF-THE-ART COMPUTER CODES ON:

- Effects of using an anisotropic layup in the attached flange
- Maximum load of stiffened panels made of brittle materials
- Change in buckle pattern

MECHANICS OF METAL MATRIX COMPOSITES:
A Program Review

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ABSTRACT

Many projected DoD and NASA vehicles of the future will require materials that can operate at high temperatures and retain significant strength and stiffness. These materials are also required to have as low a density as possible. Since many of the projected applications are at temperatures where polymer matrix materials cannot be used, metal matrix composites (MMC) are prime candidates.

In response to these anticipated requirements, the Mechanics of Materials Branch (formerly the Fatigue and Fracture Branch) in the Materials Division at NASA Langley Research Center has initiated a program to investigate the new emerging MMC by building upon our past experience with MMC materials [1-10]. This metal matrix progam is currently staffed at a 5 man-year effort in-house with three external grants or contracts. The purpose of this research is threefold: (1) Develop a complete understanding of the mechanics of MMC and their related failure modes, (2) Develop a micromechanics understanding of MMC in order to help material scientists in developing new, improved MMC materials, and (3) Develop the appropriate life-design criteria.

The current program consists of investigating both continuous and discontinuous fiber-reinforced metals. The emphasis in these material systems will be on basic material constitutive behavior, fatigue response of notched and unnotched coupons, and the static strength of notched and unnotched coupons. In addition, the influence of thermal loading will be addressed, where appropriate. Analytical models will also be developed to understand and predict all of these responses.

Experimental Program

Temperatures of approximately 1500°F are expected to occur in some of the structural components of the X-30 (the prototype National Aerospace Plane (NASP)). Emphasis is now on developing a composite material with a titanium-aluminide matrix. Such materials have not matured to the point where it is practical to conduct a full material characterization program. Therefore, an alternate material system has been chosen to serve as a model material to help identify fundamental failure mechanisms and material response of high strength titanium-based composite materials. The model system being investigated by NASA Langley is the Ti-15-3/SCS_6 composite material.

Reference [8] is a summary of the basic material response found for the Ti-15-3/SCS_6 at room temperature. It was found that the fiber/matrix interface was weak compared to the strength of the matrix and the transverse strength of the fiber. Thus, early fiber/matrix separations were observed in the off-axis plies at rather low load levels, approximately 20 ksi. These fiber/matrix separations resulted in significant reductions in the laminate stiffness. This study also presented experimental values of Young's modulus, Poisson's ratio, and static unnotched strength of five layups ($[0]_8$, $[90]_8$, $[0/90]_{2s}$, $[0/\pm45/90]_s$, and $[0_2/\pm45]_s$). In each case there was good correlation between experiment and analysis. More recent data [11] has shown the correlation between the stress level in the 0° fibers in a laminate and the experimentally determined fatigue life.

Future experimental work planned for this model material includes high temperature mechanical and fatigue characterization. The viscoelastic/viscoplastic response of the matrix material and the laminates will be examined. Further, both the fatigue behavior and static strength behavior of notched specimens will be investigated.

Whisker and particulate reinforced aluminum are being investigated to determine the effect of the discontinuous reinforcement on fatigue crack initiation and subsequent growth. The current materials include both whisker and particulate reinforced 2124 aluminum. Materials with fiber volume fractions of 0.0, 0.15 and 0.30 are being investigated in three different plate thicknesses, 0.25, 0125, and 0.07 inches. Microstructural characterization of each material condition will be completed early in the program. This program will also include a study of how the anisotropic material properties effect fracture behavior and test methods.

Dispersion strengthened titanium-aluminide (XD TiAl) composite material will also be investigated in a manner similar to that for the discontinuously reinforced aluminum.

Analytical Program

Recent analytical work [9, 10] has included a study of the best way to analyze continuous fiber reinforced MMC using the 3-D elastic-plastic finite-element code PAFAC[6]. Included in these studies was the effect of mesh geometry at a slit tip and the effect of a matrix material interlayer between laminate plies.

Our current efforts in developing the appropriate analytical tools for evaluating continuous fiber reinforced MMC are focused on modeling thermal effects with existing analytical capabilities. We have recently incorporated a thermal residual stress calculation capability into the AGLPLY [12] laminate code. Our next endeavor will be to develop time and temperature dependent analytical capabilities based upon the properties of the constituents. The capabilities will be included in the AGLPLY program, as well as the finite-element code PAFAC.

We are also using NASTRAN to develop a discrete fiber/matrix model to determine the local stress and strain distributions around a fiber. We hope to marry this approach with the AGLPLY and PAFAC codes to develop a global-local analysis to determine local microstresses in any arbitrary layup.

Currently, there seems to be no very good way to predict the mechanical response of particulate or whisker reinforced MMC apriori. We hope to be able to correlate the microstructural characterization of material systems with observed mechanical behavior. This may then serve as a basis for the development of a micromechanics model for discontinuous reinforced MMC.

REFERENCES

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[2] Dvorak, G. J; and Johnson, W. S.: "Fatigue of Metal Matrix Composites," International Journal of Fracture, Vol. 16, No. 6, Dec. 1900, pp. 585-607.

[3] Johnson, W. S.: "Mechanisms of Fatigue Damage in Boron/Aluminum Composites," Damage in Composite Materials, ASTM STP 775, K. L. Reifsnider, Ed., American Society for Testing and Materials, Philadelphia, 1982, pp. 83-102.

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MECHANICS OF METAL MATRIX COMPOSITES:
A Program Review

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OBJECTIVE

- To review the metal matrix composites program in the Mechanics of Materials Branch (formerly the Fatigue and Fracture Branch) in the Materials Division at NASA Langley.

PROGRAM OBJECTIVES

- To advance the understanding of the fatigue, fracture, and mechanical behavior of MMC's to a level which they may be safely and efficiently used on vehicles such as NASP, Hi-Star, and ATF.
- To develop an in-depth understanding of the micro-mechanical behavior of MMC's to help guide new material development.

CURRENT ACTIVITIES AND ISSUES

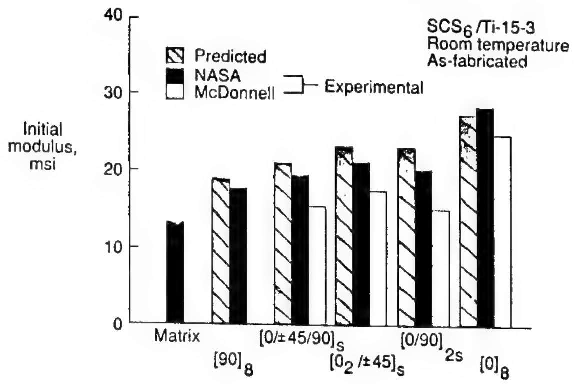
Ti-15-3 / SCS-6 Material

- Mechanical properties: Fatigue: Strength
- Damage model development

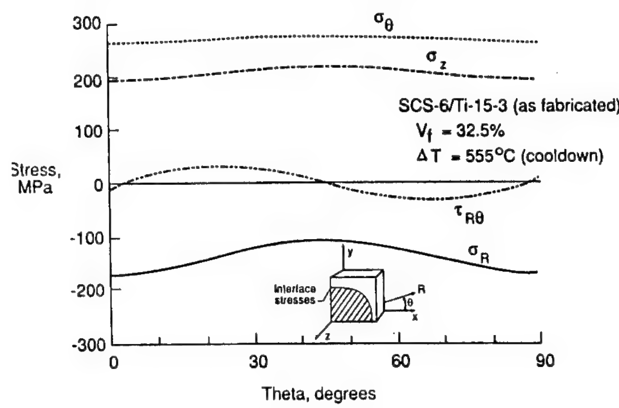
Computational activities

- PAFAC evaluate the influences of mesh geometry and matrix interlayers
- Discrete fiber/matrix modeling
- Residual thermal stress model

INITIAL MODULUS (1st CYCLE) FOR AS-FABRICATED MATERIAL



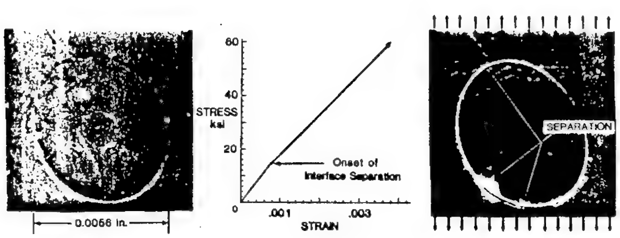
THERMALLY INDUCED RESIDUAL STRESSES



FIBER/MATRIX INTERFACE FAILURE

$SCS_6/Ti\text{-}15\text{-}3$

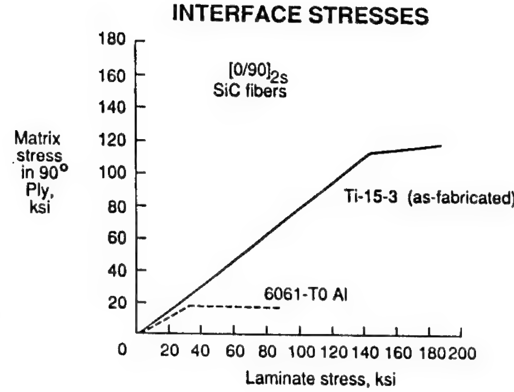
$[0/90/0/90]_s$



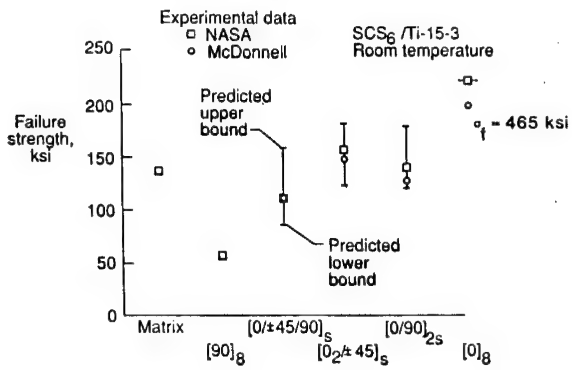
UNLOADED

60 KSI

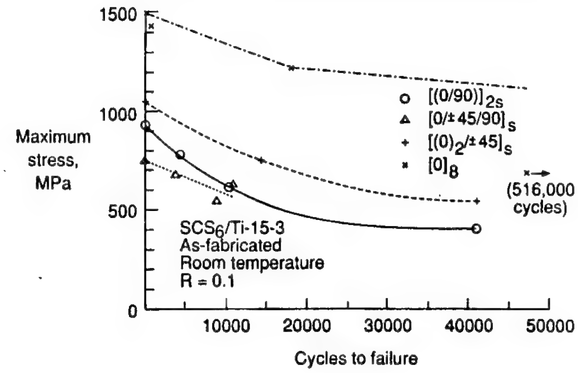
INTERFACE STRESSES



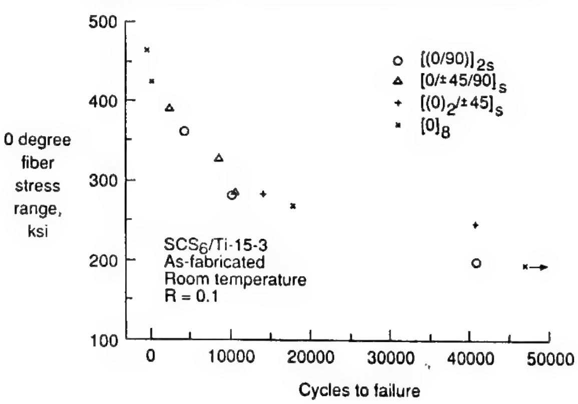
STATIC STRENGTH OF AS-FABRICATED MATERIAL



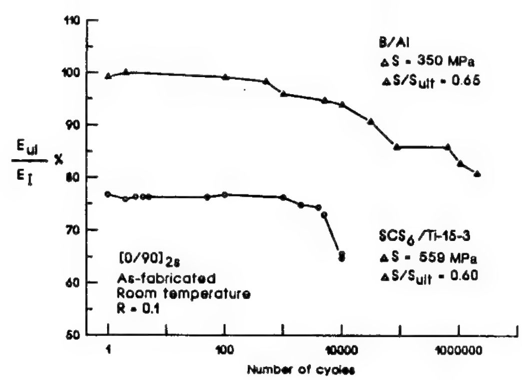
FATIGUE DATA FOR SCS-6/Ti-15-3



STRESS IN 0° FIBER DICTATES FATIGUE LIFE



STIFFNESS LOSS FOR $[0/90]_{2s}$ LAMINATES

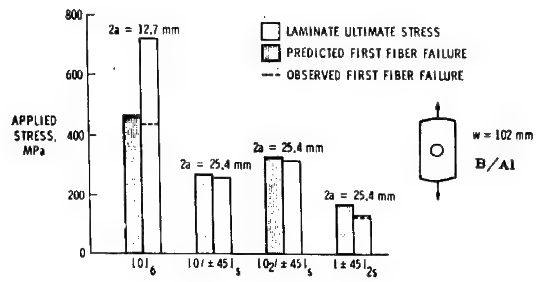


PAFAC

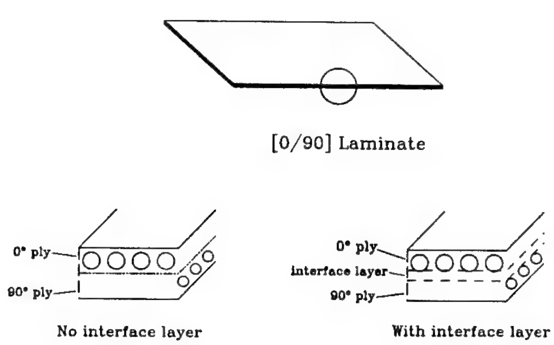
(Plastic and Failure Analysis of Composites)

- **Performs 3D elastic–plastic finite element analysis**
- **Uses a continuum materials model (VFD)**
- **Requires only the properties of constituents**
- **Models nonlinear stress–strain behavior of matrix**
- **Assumes fibers remain perfectly elastic**
- **Calculates fiber stresses in each element**
- **Incorporates fiber failure criterion based on stresses**

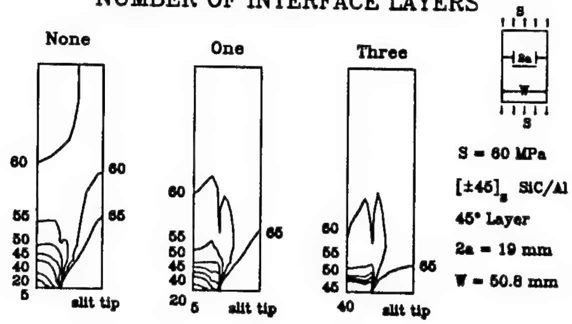
FIRST FIBER FAILURE PREDICTIONS FOR SPECIMENS WITH HOLES



MODELING OF INTERFACE LAYER



σ_y STRESS CONTOURS AS FUNCTION OF NUMBER OF INTERFACE LAYERS



All stresses are in MPa

NEAR TERM ACTIVITIES AND ISSUES

Ti-15-3 / SCS-6 Material

- Notch effects: Fracture
- [0/90/0] lay-ups (McAir)

Temperature testing

- Set up high temperature test equipment
- Test Ti-15-3 MMC

SiCp & SiCw / 2024 Al

- Micro-mechanical damage characterization
- Discontinuous reinforcement model dev.

Ti-6-4 / SCS-6 DCB Specimens (G. E.)

TiAl / SCS-6 characterization

Computational

- Develop next generation PAFAC (3D FE)
- El-pl, Visco, thermal stresses

CONTINUOUS FIBER MMC

EXPERIMENTAL PROGRAM:

Ti-15-3/SCS6: $[0]_8$, $[0/90]_{2s}$, $[0/\pm45/90]_s$, $[0_2/\pm45]_s$, and [0/90/0]

- o Thermomechanical Fatigue Characterization
- o Notched Fatigue Behavior
- o Notched Static Strength
- o Viscoelastic Characterization

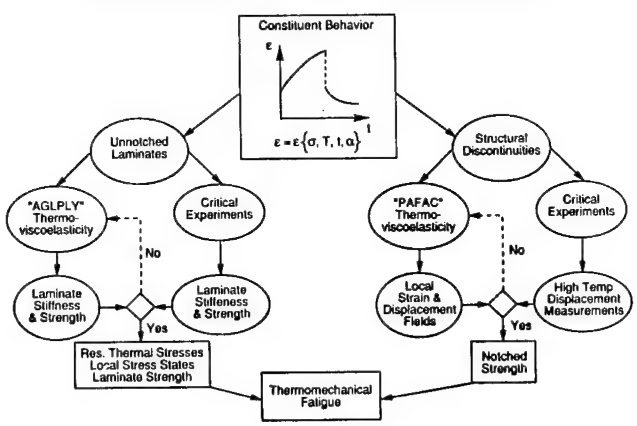
Ti-6-4/SCS6: $[0]_8$, $[90]_8$

- o Fiber/Matrix Interface Strength
- o Interface Strength Measurement

ANALYTICAL PROGRAM:

- o Thermoviscoplastic Model for Ti Matrix
- o PAFAC Program to Include Thermoviscoplasticity
- o Develop and Apply Discrete Fiber/Matrix Model to MMC
- o Determine Best Approaches and Models for Life Predictions

MMC BEHAVIOR AT ELEVATED TEMPERATURES



DISCONTINUOUS FIBER MMC

Experimental Program:

SiC_p and SiC_w/2024 Al
15% and 30% V_f
t = .25", .125", and .07"

XD TiAl

- o Mechanical Behavior
- o Fatigue Crack Initiation and Growth
- o Fracture Toughness

Analytical Program:

- o Develop Model of Fiber-Matrix Interaction
- o Develop a Model to Predict Mechanical Properties Given the Constituents and Their Orientations
- o Determine Best Approaches and Models for Life Predictions

OBJECTIVES FOR DISCONTINUOUS RENFORCED MMC

Microscopic characterization of the reinforcement

- Distribution
- Size
- Orientation

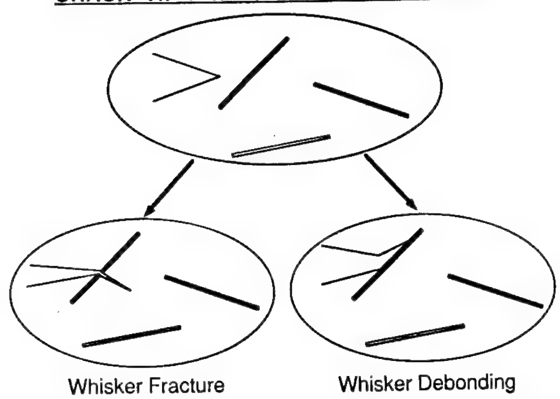
Mechanical properties

- Tensile properties
- Fracture toughness
- Fatigue resistance

Develop micromechanics models

- Material constitutive properties
- Matrix/reinforcement interactions

CRACK TIP / WHISKER INTERACTIONS



LONG TERM ACTIVITIES AND ISSUES

Fatigue and fracture of MMC joints

- Diffusion bonded (MMB)
- Mechanically fastened

New material systems

- RSR Ti MMC
- XD
- AMMC

METCAN -- THE METAL MATRIX COMPOSITE ANALYZER

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Pappu L. N. Murthy
NASA Resident Research Associate (Cleveland State University)

Dale A. Hopkins
NASA Lewis Research Center

ABSTRACT

Metal matrix composites (MMC) have been the subject of intensive study recently and are receiving serious consideration for critical structural applications in advanced aerospace systems. The routine application of MMC in aerospace structures will evolve as concurrent developments progress in related areas of processing and fabrication, experimental mechanics, and computational structural analysis and design methodologies. This presentation concerns recent research efforts related to the latter aspect, namely, MMC analysis and design.

Predicting the mechanical and thermal behavior and the structural response of components fabricated from MMC requires the use of a variety of mathematical models. These models, for example, relate stresses to applied forces, stress intensities at the tips of cracks to nominal stresses, buckling resistance to applied force, or vibration response to excitation forces. The models just mentioned require initial tangent and strain-dependent stress-strain relationships. Experimental data indicate that the stress-strain response of unidirectional MMC are: (1) slightly nonlinear in the longitudinal direction, (2) mildly nonlinear in the transverse direction, and (3) highly nonlinear in intralaminar shear. In-service loads on MMC structures can generally be expected to strain the metal matrix nonlinearly. The stress-strain relationships for a laminate may become load-path dependent, and hence it is important to be able to track the composite behavior throughout its load history. Moreover, the mechanical performance and structural integrity of MMC are ultimately governed by the behavior of the constituents at a local level. In general, this behavior is dynamic because of various nonlinearities associated with, for example: (1) large local stress excursions, (2) temperature-dependent material properties, (3) time-dependent effects, and (4) constituent chemical reaction. It is important also then to be able to track behavior at the local level.

Extensive research in computational mechanics methods for predicting the nonlinear behavior of MMC. This research has culminated in the development of the METCAN (Metal matrix composite analyzer) computer code. The presentation described METCAN and includes typical results to illustrate its capability to simulate the history of unidirectional MMC nonlinear behavior starting with the fabrication process, cooled down to room temperature and subjected to thermal and mechanical cyclic loadings.

Specifically, the history results include: (1) stress-strain curves (longitudinal and transverse), (2) microstress, (3) moduli, (4) thermal expansion coefficients, and (5) thermal heat conductivities. Included also is a summary of METCAN present capability and near-future conditions.

METCAN—THE METAL-MATRIX COMPOSITE ANALYZER

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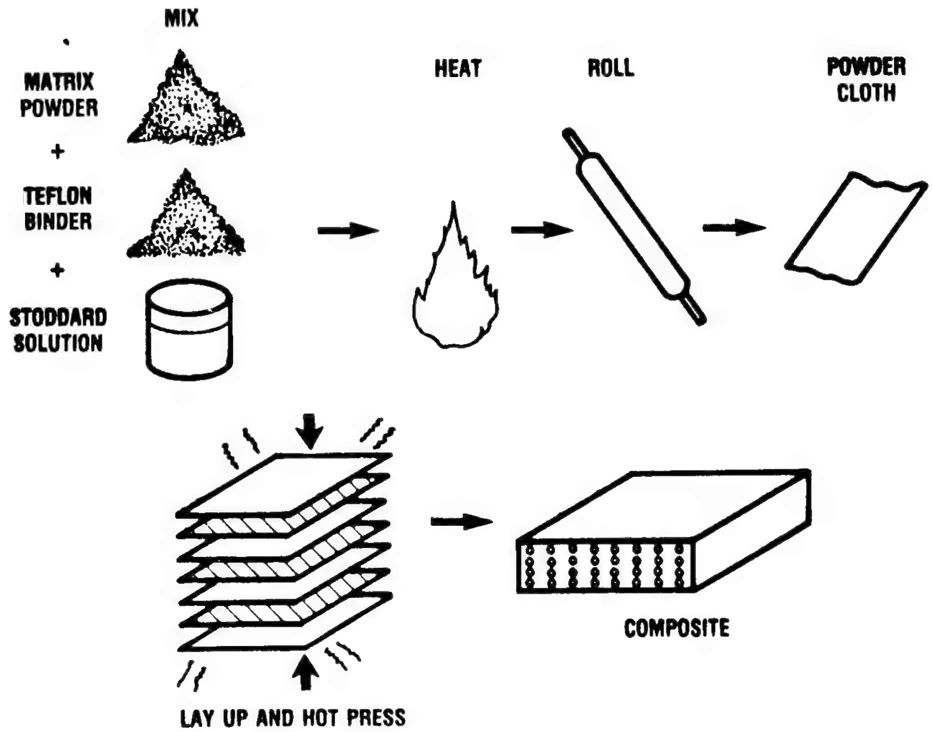
CD-88-32971

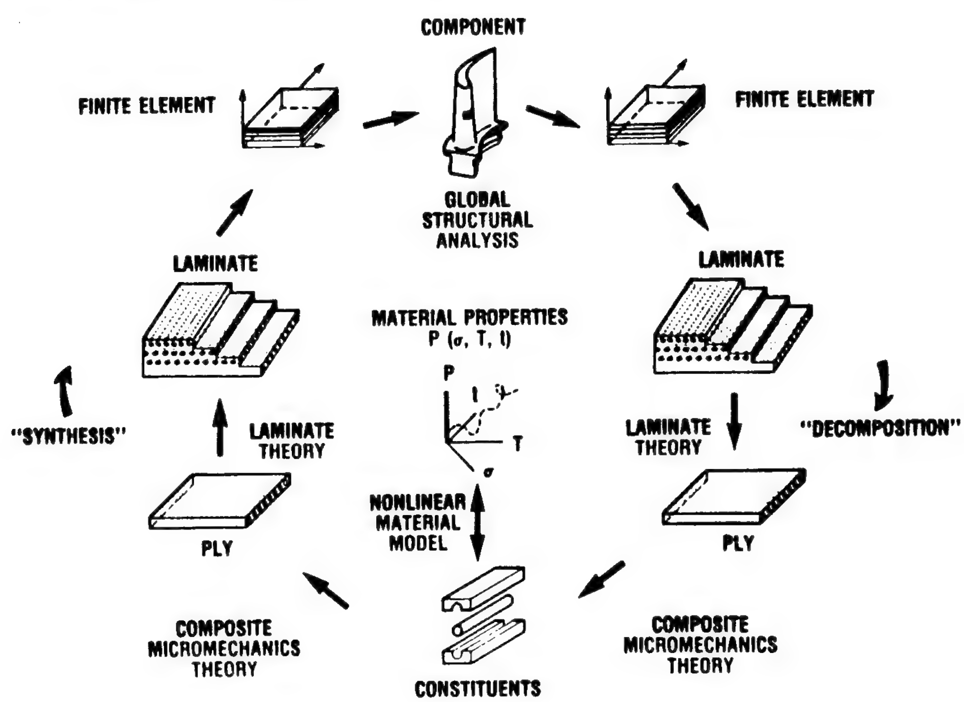
OVERVIEW OF STRUCTURAL ANALYSIS

SYSTEM EQUATIONS OF MOTION	$[M]\{\ddot{u}\} + [C]\{\dot{u}\} + [K]\{u\} = \{F(t)\}$ $\searrow \{u\} \le \{u_A\}$
CONSTITUTIVE EQUATION	$\{\sigma\} = [D][B]\{u\}$ $\searrow \{\sigma\} \le \{S_A\}$
NATURAL FREQUENCY EIGENPROBLEM	$\langle [K] - \omega^2[M] \rangle \{u\} = \{o\}$ $\searrow \omega \le \omega_A$

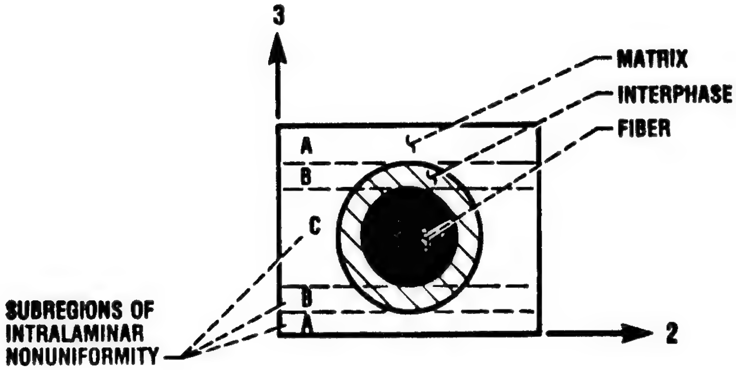
METAL-MATRIX COMPOSITE FABRICATION PROCESS



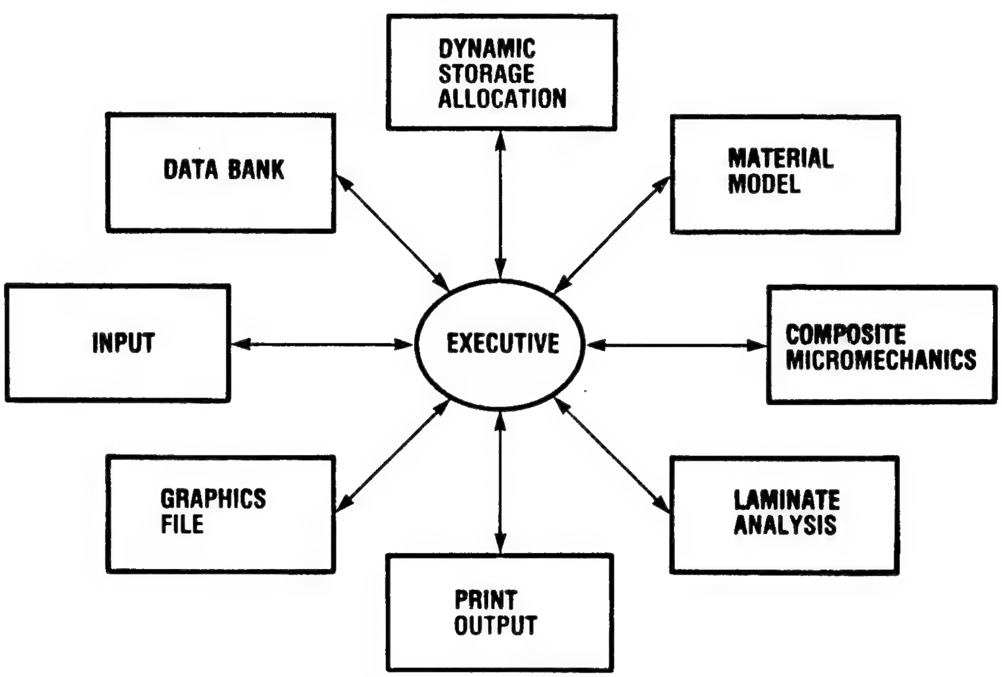
INTEGRATED APPROACH TO METAL-MATRIX COMPOSITE ANALYSIS



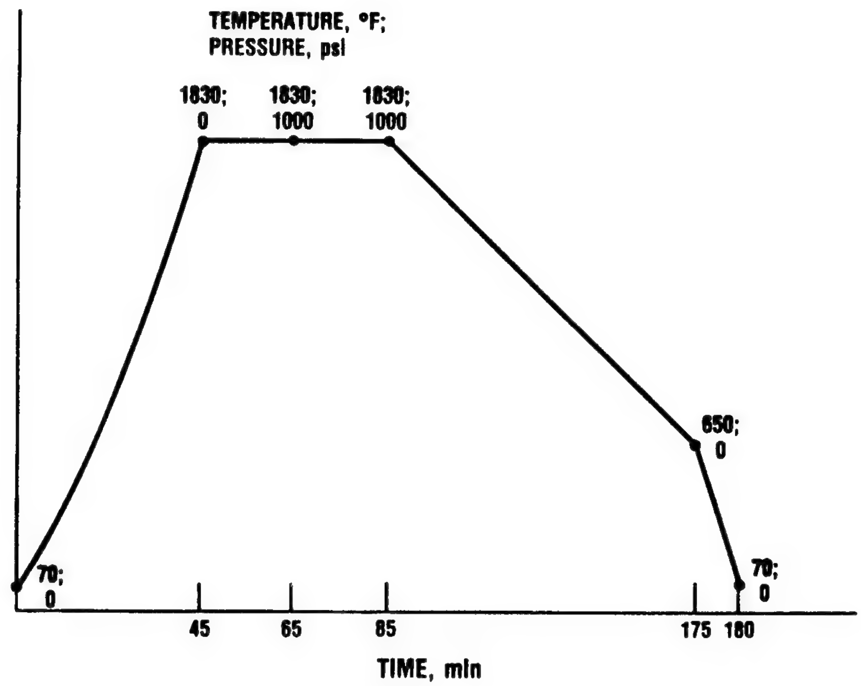
COMPOSITE LOCAL BEHAVIOR AND RESPONSE



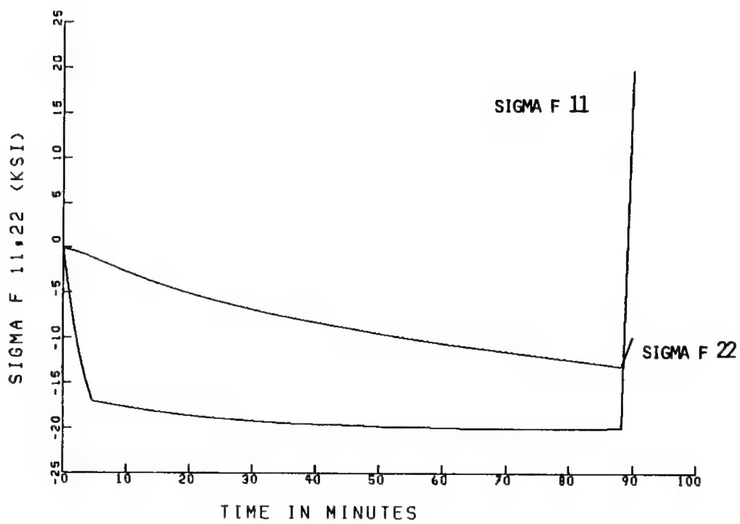
MODULAR STRUCTURE OF METCAN



MMC PROCESSING CONDITIONS

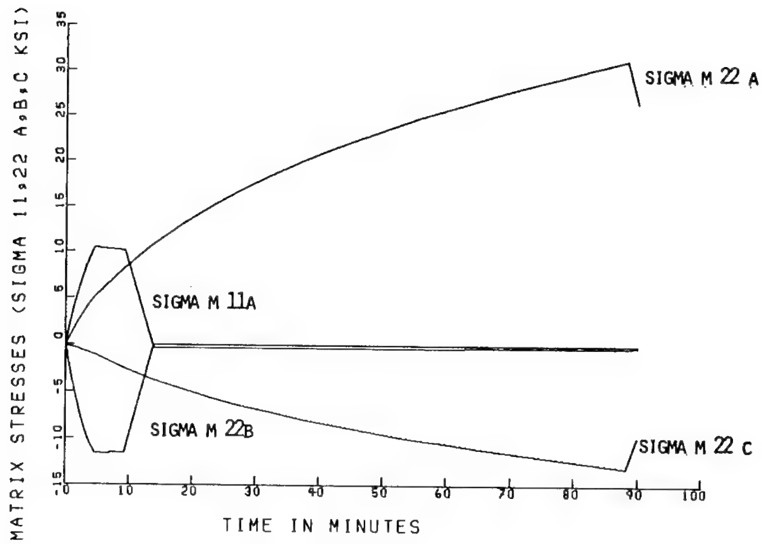


[0_4] P100/COPR WITH 5% INTERFACE
FIBER MICROSTRESSES



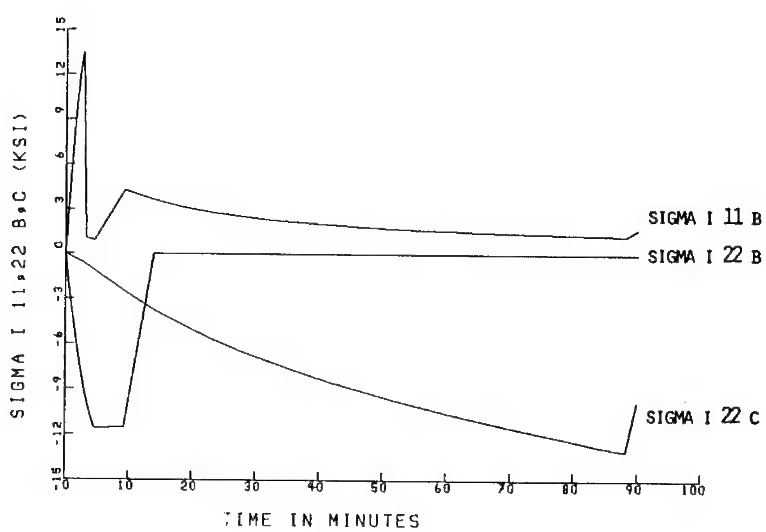
UNIDIRECTIONAL P100/COPR WITH 5% INTERFACE

MATRIX MICROSTRESSES



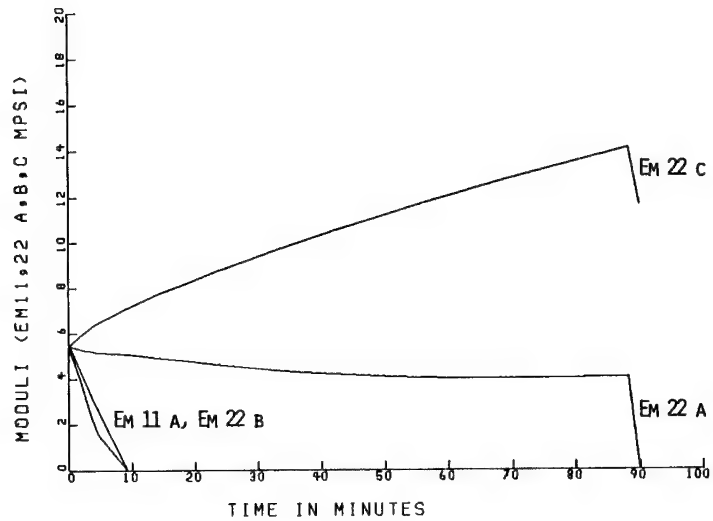
UNIDIRECTIONAL P100/COPR WITH 5% INTERFACE

INTERFACE MICROSTRESSES



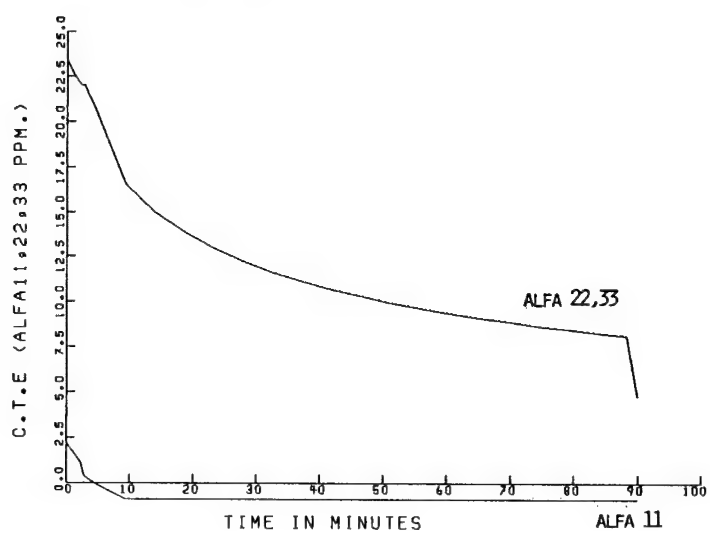
UNIDIRECTIONAL P100/COPR WITH 5% INTERFACE

MATRIX MODULI



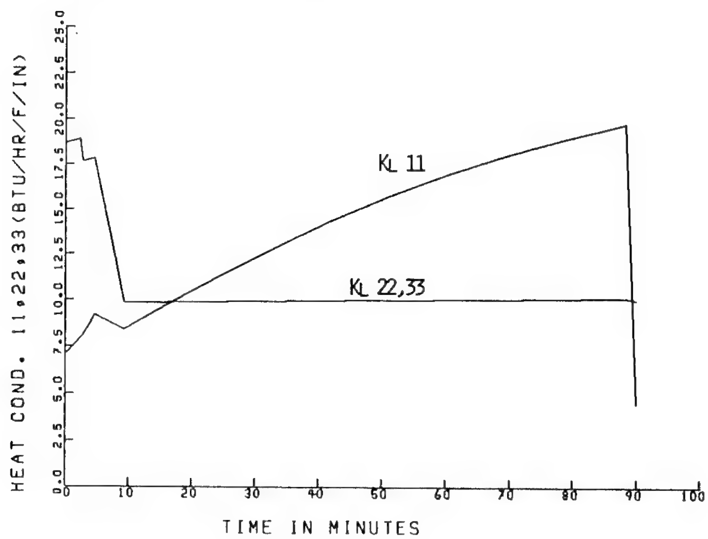
UNIDIRECTIONAL P100/COPR WITH 5% INTERFACE

PLY THERMAL EXP. COEFF.



UNIDIRECTIONAL P100/COPR WITH 5% INTERFACE

PLY THERMAL CONDUCTIVITIES



METCAN CURRENT AND FUTURE* CAPABILITIES

- **SIMULATED BEHAVIOR AND RESPONSE**
  - MECHANICAL PROPERTIES
  - THERMAL PROPERTIES
  - STRESSES AND STRAINS
  - MICROSTRESSES
  - FRACTURE STRESSES
  - FINITE ELEMENT COMPATIBILITY
- **LOAD AND EFFECT**
  - MONOTONIC LOAD HISTORIES
  - CYCLIC LOAD HISTORIES
  - * INTERPHASE PROPERTIES AND GROWTH
  - * STEADY-STATE CREEP
  - * THERMAL RATCHETTING
  - * CREEP RUPTURE
  - * DAMAGE
  - * THERMAL SHOCK
  - * IMPACT LOADING

SUMMARY

- METCAN PERFORMS MOST ESSENTIAL ASPECTS OF MECHANICS, ANALYSIS, AND DESIGN OF METAL-MATRIX COMPOSITES
- METCAN IS MODULAR, OPEN ENDED, AND USER FRIENDLY
- STANDARD METAL-MATRIX COMPOSITES AS WELL AS INTERPLY HYBRID METAL-MATRIX COMPOSITES CAN BE ANALYZED
- RESPONSE DUE TO DIFFERENT TYPES OF THERMAL AND MECHANICAL LOAD HISTORIES ACCOUNTING FOR THERMOMECHANICAL DEGRADATION CAN BE OBTAINED ROUTINELY
- KEY FEATURES OF METCAN INCLUDE
  - LINEAR AND NONLINEAR ANALYSIS
  - ROOM- AND HIGH-TEMPERATURE PROPERTIES
  - STRESS AND STRAIN INFLUENCE COEFFICIENTS
  - RESIDENT DATA BANK

EFFECTS OF CUTOUTS ON THE BUCKLING OF COMPOSITE PLATES

by

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ABSTRACT

In recent years, research efforts have focused on innovative structural analysis and design of high performance aerospace vehicles. The long-range goal of using advanced composite materials in the primary structure of these vehicles has led to the search for ways to exploit their properties. In addition to high strength and stiffness, this class of materials offers an added degree of freedom for tailoring structural response by altering fiber orientation and stacking sequence of the layers comprising the structural component which is not offered by metals.

An important subcomponent used in aerospace vehicles is the rectangular plate with a central circular cutout. Cutouts commonly appear in rectangular panels as access ports for mechanical and electrical systems, or are included to reduce the vehicle weight. Often during flight, these members experience compression loads, and thus their ability to resist buckling is important in design.

This paper presents the simple and inexpensive approximate analysis for accurately predicting buckling loads and trends of compression-loaded specially-orthotropic plates with a centrally located cutout described in reference 1. The applicability of this analysis to symmetrically-laminated angle-ply plates is also discussed [2,3]. Results of a parametric study, and experimental results, are presented that show the buckling resistance of a plate is not always reduced by increasing the cutout size, and that buckling resistance can be substantially improved by optimally selecting cutout size, fiber orientation, and stacking sequence of a plate. Results are also presented that indicate the importance of prebuckling load distribution and loss of bending stiffness associated with the cutout, plate aspect ratio, and boundary conditions on buckling resistance.

REFERENCES

1. M. P. Nemeth, M. Stein, and E. R. Johnson, "An Approximate Buckling Analysis for Rectangular Orthotropic Plates With Centrally Located Cutouts," NASA Technical Paper 2528, 1986.

2. M. P. Nemeth, "Importance of Anisotropy on Buckling of Compression-Loaded Symmetric Composite Plates," AIAA Journal, Vol. 24, No. 11, November 1986, pp. 1831-1835.

3. M. P. Nemeth, "Buckling Behavior of Compression-Loaded Symmetrically Laminated Angle-Ply Plates with Holes," AIAA Journal, Vol. 26, No. 3, March 1988, pp. 330-336.

EFFECTS OF CUTOUTS ON THE BUCKLING OF COMPOSITE PLATES

by

Michael P. Nemeth
NASA Langley Research Center
for the

Thirteenth Annual Mechanics Of Composites Review

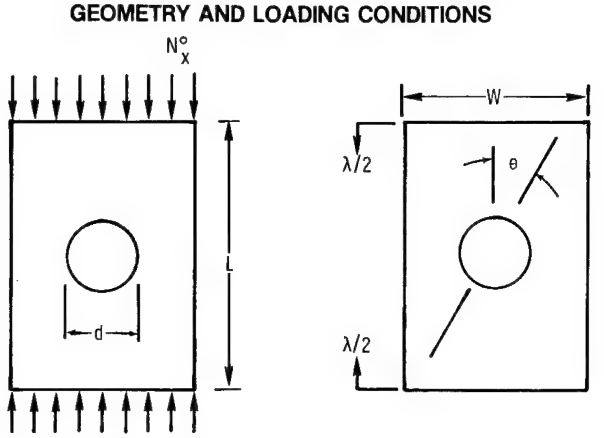
November 2-3, 1988

Bal Harbour, Florida

OBJECTIVES

- To present a simple and inexpensive analysis for predicting buckling of orthotropic plates with centrally located cutouts
- To present the results of parametric study of buckling of composite plates with central circular cutouts

GEOMETRY AND LOADING CONDITIONS



a. Stress loading

b. Displacement loading

ANALYSIS ASSUMPTIONS

- Rectangular planform
- Centrally located symmetrical cutouts
- Displacement and stress loadings
- Simply supported unloaded edges
- Clamped or simply supported loaded edges
- Isotropic and specially - orthotropic laminates with uniform thickness

ANALYTICAL FORMULATION

PREBUCKLING

- Approximate the inplane displacements

$$\begin{Bmatrix} u \\ v \end{Bmatrix} = \begin{Bmatrix} u_0(x) \\ v_0 y \end{Bmatrix} + \sum_{k=1}^{N} \begin{Bmatrix} u_{2k-1}(x) \cos (2k-1)\frac{\pi y}{2b} \\ v_{2k-1}(x) \sin (2k-1)\frac{\pi y}{2b} \end{Bmatrix}$$

- Reduce potential energy functional to 1-D
- Obtain 1-D system of ordinary differential equations
- Solve numerically

ANALYTICAL FORMULATION

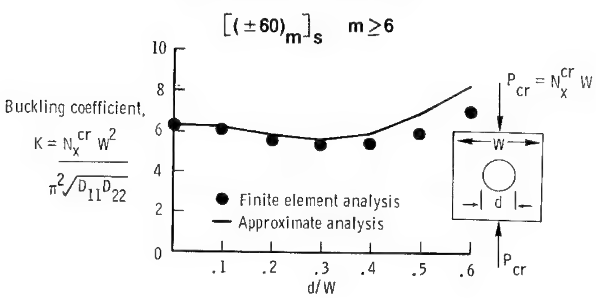
BUCKLING

- Approximate the out-of-plane displacements

$$w = \sum_{k=1}^{S} w_{2k-1}(x) \cos (2k-1)\frac{\pi y}{2b}$$

- Reduce second variation of potential energy functional to 1-D
- Obtain 1-D system of ordinary differential equations with variable coefficients
- Obtain algebraic eigenvalue problem
- Solve numerically

COMPARISON WITH FINITE ELEMENT RESULTS
SIMPLY SUPPORTED PLATES



TEST FIXTURE AND SPECIMEN

AS4/3502

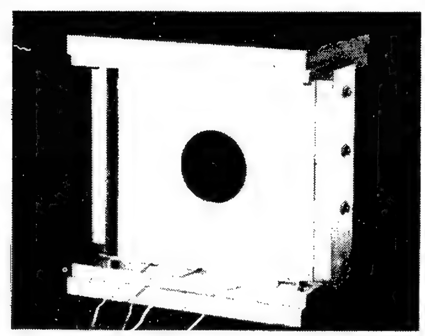
$[(\pm 30)_6]_s$

$[(\pm 60)_6]_s$

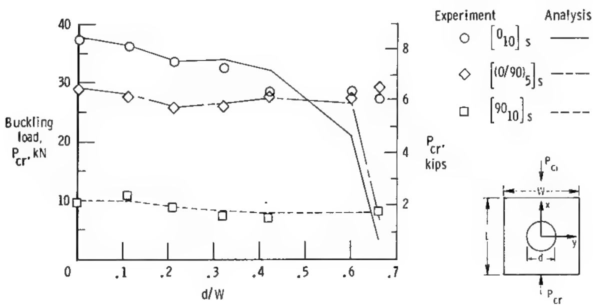
$[0_{10}]_s$

$[90_{10}]_s$

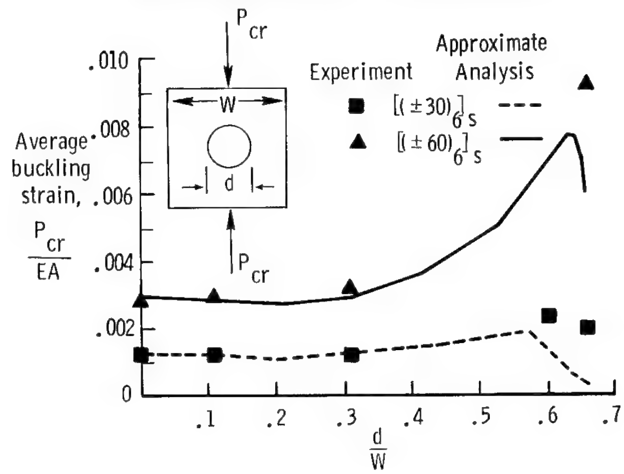
$[(0/90)_5]_s$



COMPARISON WITH EXPERIMENTAL RESULTS

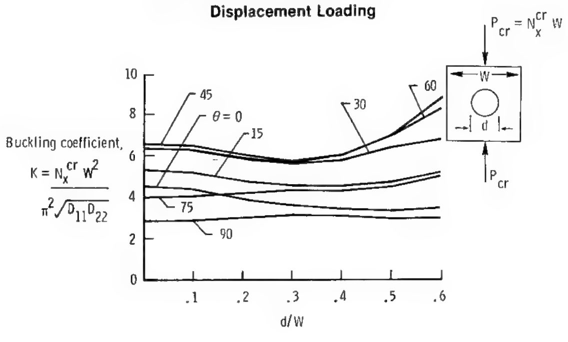


COMPARISON WITH EXPERIMENTAL RESULTS



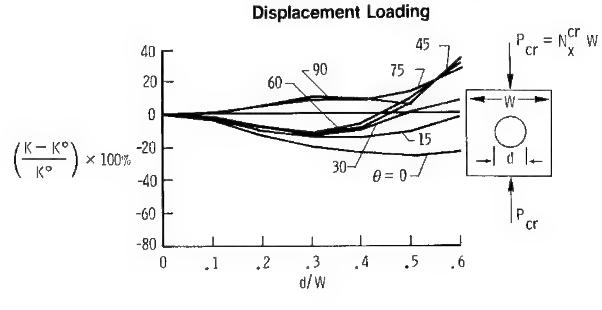
SQUARE SIMPLY SUPPORTED PLATES

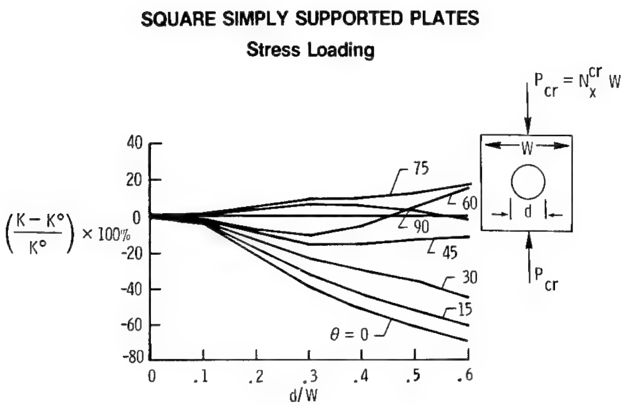
Displacement Loading



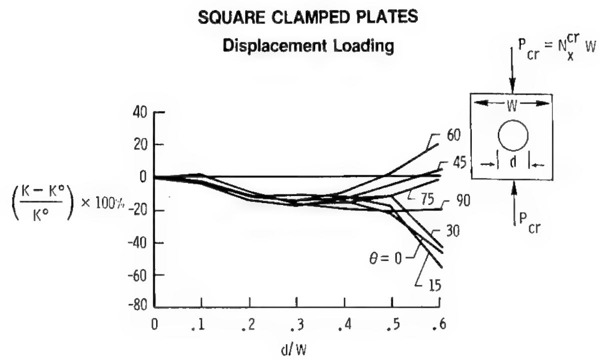
SQUARE SIMPLY SUPPORTED PLATES

Displacement Loading

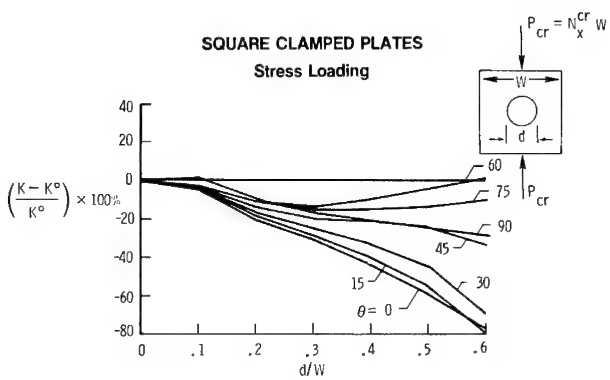




SQUARE SIMPLY SUPPORTED PLATES
Stress Loading
$P_{cr} = N_x^{cr} W$
W
d
P_{cr}
$\left(\frac{K - K°}{K°}\right) \times 100\%$
d/W
75
60
90
45
30
15
$\theta = 0$



SQUARE CLAMPED PLATES
Displacement Loading
$P_{cr} = N_x^{cr} W$
W
d
P_{cr}
$\left(\frac{K - K°}{K°}\right) \times 100\%$
d/W
60
45
75
90
30
$\theta = 0$
15

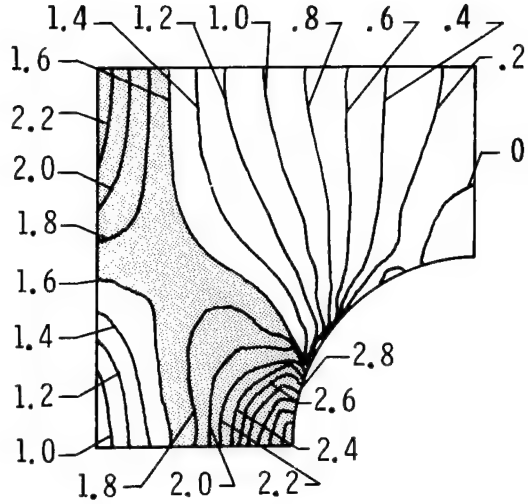


SQUARE CLAMPED PLATES
Stress Loading
$P_{cr} = N_x^{cr} W$
W
d
P_{cr}
$\left(\frac{K - K°}{K°}\right) \times 100\%$
d/W
60
75
90
45
15
30
$\theta = 0$

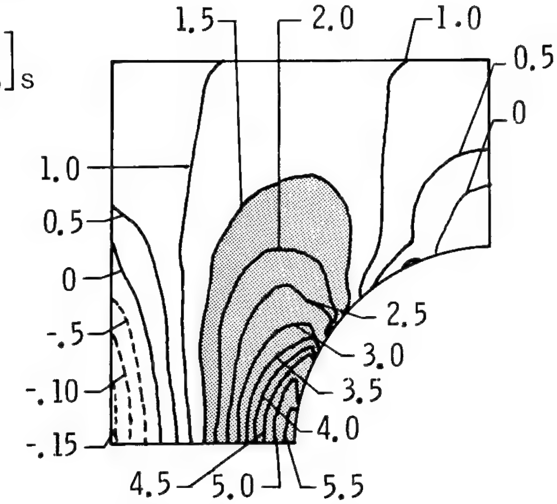
NORMALIZED STRESS RESULTANT CONTOURS

Nx Contours

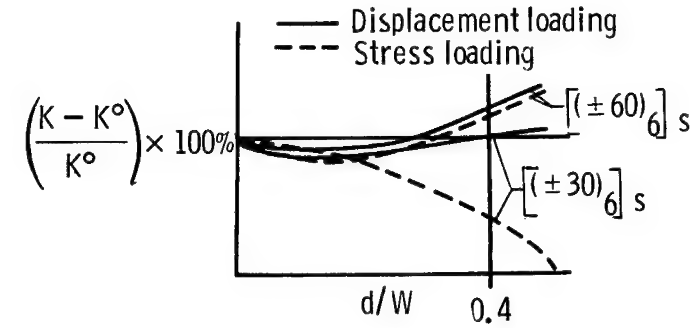
$[(\pm 30)_6]_s$



(a) Displacement loading

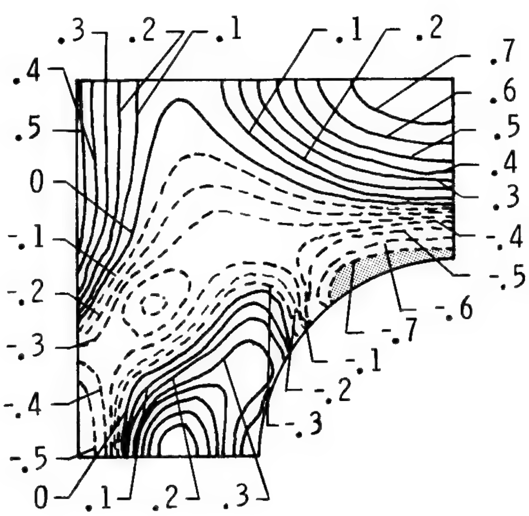


(b) Stress loading

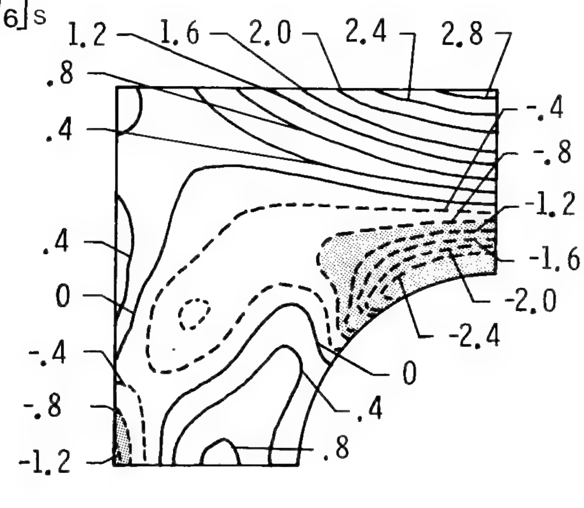


Ny Contours

$[(\pm 60)_6]_s$

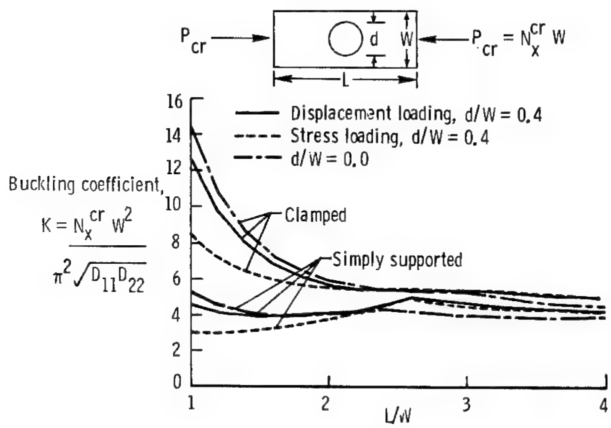


(a) Displacement loading

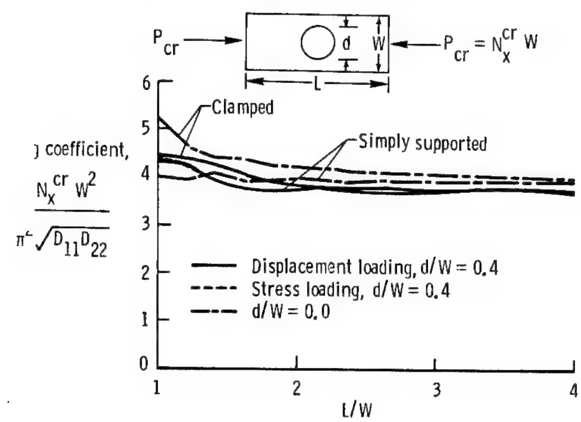


(b) Stress loading

RESULTS FOR $[(\pm 15)_m]_s$ PLATES



RESULTS FOR $[(\pm 75)_m]_s$ PLATES



CONCLUDING REMARKS

- Approximate analysis accurately predicts buckling loads and trends
- Analysis and experimental results show a trend of increasing buckling load with increasing hole size for some plates
- Buckling behavior depends on loss in bending stiffness and inplane load distribution due to the hole
- Buckling behavior of rectangular plates depends on change in buckle aspect ratio and inplane load distribution due to the hole
- Effect of the hole diminishes as the plate aspect ratio increases
- Differences in buckling load due to to differences in loading and boundary conditions attenuate faster as θ approaches 90 degrees

MICROMECHANICS FOR FIBER COMPOSITE DAMPING

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ABSTRACT

The significance of material damping to the dynamic performance of structures is broadly recognized. Passive damping controls the dissipation of the energy induced into a structure, and has been proved to be an important design parameter for vibration control, fatigue endurance, and impact resistance. It is well known that fiber/polymer-matrix composites may provide one or two orders of magnitude higher material damping than common metals, a factor that makes them further attractive as structural materials, since they may simultaneously provide superior elastic properties and passive damping capacity. An additional appealing design factor is the possibility to tailor the composite damping by controlling the anisotropy of the composite material. In order to realize significant structural benefits from the inherent damping capacity of composite materials, suitable mechanics theories should be formulated which will analytically correlate the damping of a composite structure to the properties of the basic constituents, ply stacking sequence, hygro-thermal conditions, existing damage, and structural configurations. Research at Lewis Center has led to the development of an integrated theory for composite mechanics from micromechanics to structural analysis including the hygro-thermal (moisture, temperature) environmental effects [1]. Presently this research is being extended to incorporate prediction of damping. The purpose of this paper is to describe a micromechanics theory for predicting the on-axis and off-axis damping capacity of composite laminae (plies).

Approximate micromechanics equations are derived based on hysteretic damping. The equations are in explicit form where the specific damping capacity (SDC) is represented in terms of: (1) elastic and dissipative properties of the fibers and matrix, (2) interface properties, and (3) temperature and moisture. The interfacial damping is further expressed in terms of broken fibers, radial stress, and local fiber/matrix friction coefficients. The analysis includes six different damping coefficients each one corresponding to the following local stresses: (1) longitudinal, (2) transverse, (3) through the thickness normal stress, (4) in-plane shear, (5) through the thickness (x-z) shear, and (6) through the thickness (y-z) shear. Good correlation is obtained between analytical predictions and limited experimental data [2]. Results indicate that the longitudinal normal SDC significantly depends on elastic and dissipative properties of the fibers. The transverse normal SDC and the shear in-plane and out-of-plane SDC's are mainly controlled by the dissipative properties of the matrix and the elastic properties of the fibers. The results also illustrate the important effect of temperature and moisture on composite damping.

Off-axis SDC's, i.e., the SDC's of a ply loaded at an off-axis angle, are also synthesized from the on-axis SDC's. The transformations for this purpose are also presented. Obtained results illustrate wide variation of the SDC with respect to the fiber orientation and coupling between various deformation modes. Finally, the problem of temperature increase within a vibrating ply due to strain energy dissipation is addressed. Temperature distributions through the thickness of plies subjected to cyclic vibration are predicted based on Fourier's heat transfer law.

REFERENCES

1. Murthy P. L. N., and Chamis C. C., "ICAN: Integrated Composite Analyzer," AIAA Paper 84-0974, May 1984.

2. Adams R. D., Fox M. A. O., Flood R. J. L., Friend R. J., and Hewitt R. L., "The Dynamic Properties of Unidirectional Carbon and Glass Fibre-Reinforced Plastics in Torsion and Flexure," Journal of Composite Materials, Vol. 3, 1969, pp. 594-603.

MICROMECHANICS FOR FIBER COMPOSITE DAMPING

BY

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PRESENTATION OUTLINE:

- o BACKGROUND
- o COMPOSITE STRUCTURAL PERFORMANCE EVALUATION SUMMARY
- o OBJECTIVE
- o APPROACH
- o COMPOSITE MICROMECHANICS FOR DAMPING
  - FUNDAMENTALS
- o PLY DAMPING SPECIFIC DAMPING CAPACITIES (SDC) - EQUATIONS AND RESULTS
- o TEMPERATURE RISE - EQUATIONS AND RESULTS
- o NEAR-FUTURE DIRECTIONS
- o CONCLUSIONS

BACKGROUND:

- o <u>IMPORTANCE OF DAMPING IN STRUCTURAL PERFORMANCE</u>
- o DAMPING IS A VERY IMPORTANT DESIGN CONSIDERATION IN FIBER COMPOSITE STRUCTURAL PERFORMANCE UNDER CYCLIC AND IMPACT LOADING CONDITIONS. CONTROLS THE CONVERSION OF INDUCED STRAIN ENERGY TO THERMAL
- o FIBER COMPOSITES BY THEIR INHERENT HETEROGENEOUS NATURE HAVE MATERIAL DAMPING WHICH IS AT LEAST ONE ORDER OF MAGNITUDE GREATER COMPARED TO METALS
- o <u>PRESENT APPROACH TO EVALUATE DAMPING.</u>
  GENERALLY FIBER COMPOSITE DAMPING IS DETERMINED EXPERIMENTALLY

BACKGROUND (CONT'D)

- o SHORTCOMING OF PRESENT APPROACH

  EXPERIMENTAL EVALUATION IS NOT ADAPTABLE TO MULTITUDE OF CONDITIONS THAT FIBER COMPOSITES STRUCTURES CAN BE EXPOSED

- o ALTERNATE APPROACH

  COMPUTATIONAL SIMULATION PROVIDES THE FLEXIBILITY TO EVALUATE DAMPING UNDER ALL ANTICIPATED CONDITIONS

- o COMPUTATIONAL SIMULATION NEEDS MATHEMATICAL MODELS TO REPRESENT THE PHYSICS OF FIBER COMPOSITE DAMPING FROM MICROMECHANICS TO DYNAMIC STRUCTURAL RESPONSE

OBJECTIVE:

DESCRIBE AN ONGOING RESEARCH ACTIVITY TO DEVELOP MICROMECHANICS FOR FIBER COMPOSITE DAMPING, PRESENT TYPICAL RESULTS, AND OUTLINE NEAR-FUTURE DIRECTIONS

APPROACH:

- o AUGMENT AVAILABLE MICROMECHANICS FOR FIBER COMPOSITE HYGROTHERMOMECHANICAL BEHAVIOR TO ACCOUNT FOR DAMPING

- o INTEGRATE THE AUGMENTED MICROMECHANICS INTO ICAN (INTEGRATED COMPOSITE ANALYZER) COMPUTER CODE TO PREDICT PLY DAMPING

COMPOSITE STRUCTURAL PERFORMANCE EVALUATION SUMMARY

STRUCTURAL ANALYSIS MODEL (SAM)

$$[M]\{\ddot{u}\} + [C]\{\dot{u}\} + [K]\{u\} = \{F(t)\}$$

WHERE:

$$[M], [C], [K] = \mathcal{F}(x_i, T, M, t, (E, \rho, \sigma, \psi)_{f,m})$$

$$\{F\} = \mathcal{F}(x_i, F_{m,T,M})$$

SOLUTION OF SAM : $u, \omega, P_{cr}, \sigma, G, (\frac{\Delta a}{N})$

- STRUCTURAL INTEGRITY
- FATIGUE AND LIFE
- STRUCTURAL DURABILITY
- STRUCTURAL RELIABILITY

HYGROTHERMOMECHANICAL THEORY STATUS

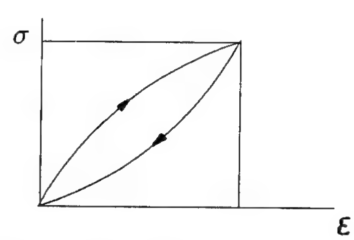
POLYMER CONTROLLED PROPERTIES:

- MECHANICAL $\frac{P_M}{P_o} = [\frac{T_{gw} - T}{T_{gd} - T_o}]^{0.5}$
- THERMAL/DAMPING $\frac{P_T}{P_o} = [\frac{T_{gd} - T_o}{T_{gw} - T}]^{0.5}$
- HYGRO $T_{gw} = T_{gd}(0.005M_l^2 - 0.1M_l + 1)$
- MOISTURE $M_l = M_m(s\beta_m k_m + k_v)\frac{\rho_m}{\rho_l}$, $M_m = M\infty\ (RHR)$
- LIFE/DURABILITY $\frac{S_N}{S_o} = [\frac{T_{gw} - T}{T_{gd} - T_o}]^{0.5} - 0.1 log N$

COMPOSITE MICROMECHANICS FOR DAMPING

FUNDAMENTAL EQUATION:

$$E_H = E_K \psi \omega$$



E_H = HEAT ENERGY GENERATED

E_K = KINETIC ENERGY PER CYCLE

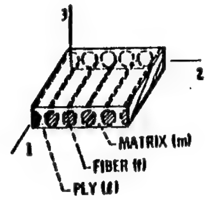
ψ = SPECIFIC DAMPING CAPACITY (SDC)

ω = FREQUENCY

PLY DAMPING

$$\psi_{l11} = \psi_{f11} k_f \frac{E_{f11}}{E_{l11}} + \psi_{mn} k_m \frac{E_m}{E_{l11}}$$

$$+ 4\sqrt{\pi k_f} N_{fb} \mu \frac{\sigma_{in}}{\sigma_{l11}} \frac{\delta_l}{t_l}$$



μ = FRICTION COEFFICIENT (FIBER PULL–OUT)

N_{fb} = NUMBER OF BROKEN FIBERS

δ_l = BROKEN FIBER INEFFECTIVE LENGTH IN R.V.

$$\delta_l = 1.15 d_f [\frac{1 - \sqrt{k_f}}{\sqrt{k_f}} (\frac{E_{f11}}{G_m})]^{0.5}$$

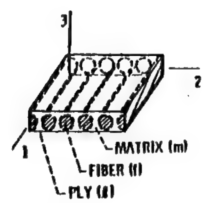
σ_{in} = INTERFACIAL STRESS ON FIBER CAUSING FRICTION

PLY DAMPING (CONT'D)

$$\psi_{l22} = \psi_{l33} = \psi_{f22} \sqrt{k_f} \frac{E_{22}}{E_{f22}} + \psi_{mn} (1 - \sqrt{k_f}) \frac{E_{22}}{E_m}$$

$$\psi_{l13} = \psi_{l12} = \psi_{f12} \sqrt{k_f} \frac{G_{12}}{G_{f12}} + \psi_{ms} (1 - \sqrt{k_f}) \frac{G_{12}}{G_m}$$

$$\psi_{l23} = \psi_{f23} \sqrt{k_f} \frac{G_{l23}}{G_{f23}} + \psi_{ms} (1 - \sqrt{k_f}) \frac{G_{l23}}{G_m}$$



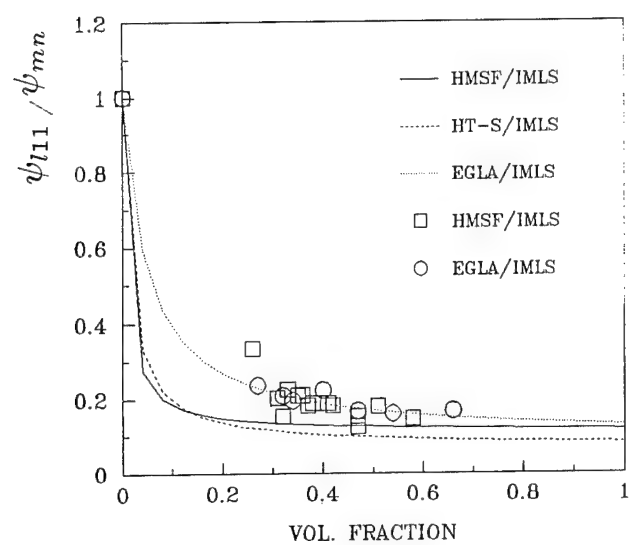
where,

$$E_{22} = (1 - \sqrt{k_f}) E_m + \frac{\sqrt{k_f} E_m}{1 - \sqrt{k_f}(1 - \frac{E_m}{E_{f22}})} \qquad G_{l23} = \frac{E_{l22}}{2(1 + \nu_{l23})}$$

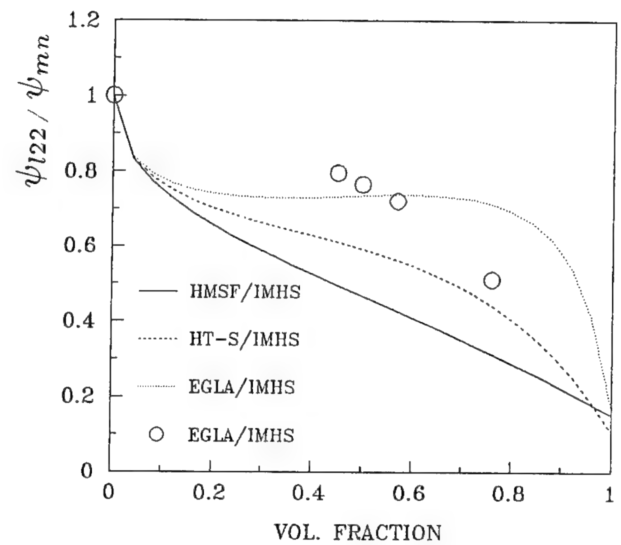
$$G_{12} = (1 - \sqrt{k_f}) G_m + \frac{\sqrt{k_f} G_m}{1 - \sqrt{k_f}(1 - \frac{G_m}{G_{f12}})} \qquad E_{l22} = \frac{E_m}{1 - \sqrt{k_f}(1 - \frac{E_m}{E_{f22}})}$$

$$\nu_{l23} = \frac{(1 - k_f)\nu_m}{1 - k_f \nu_m} (1 - k_f) + k_f \nu_{f23}$$

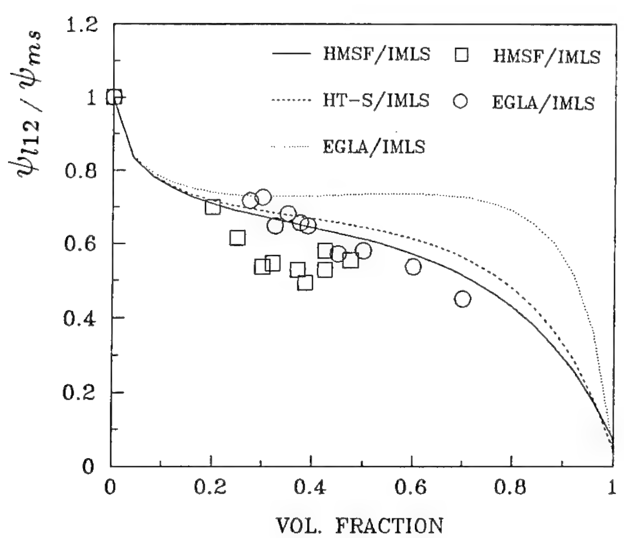
LONGITUDINAL NORMAL DAMPING



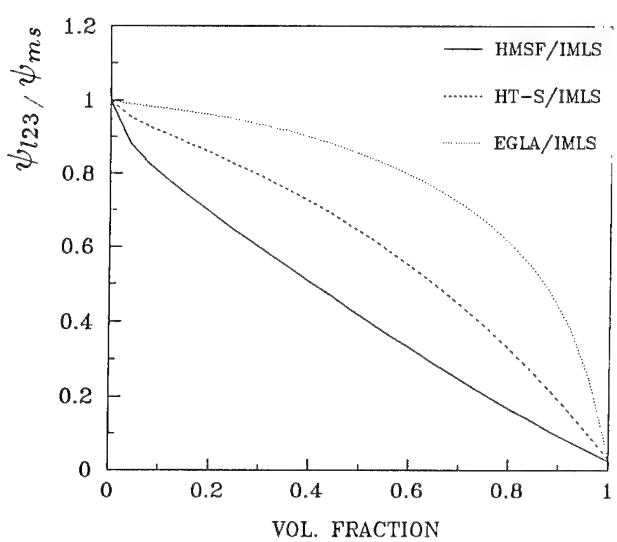
TRANSVERSE NORMAL DAMPING

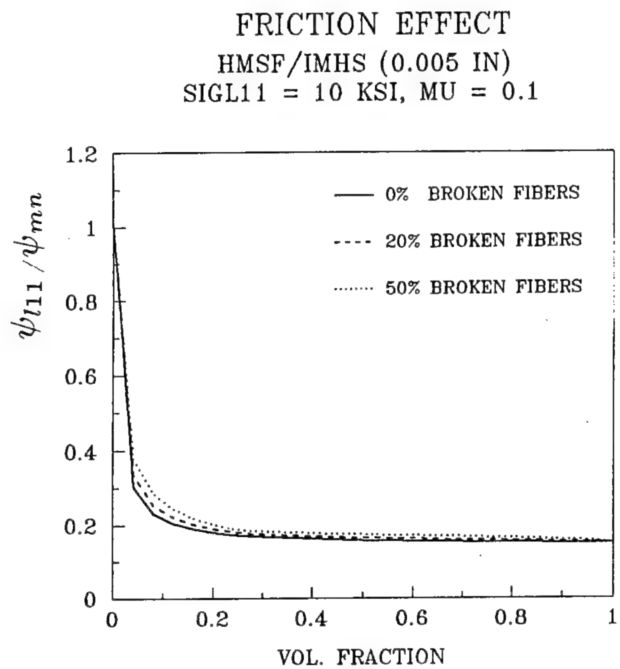


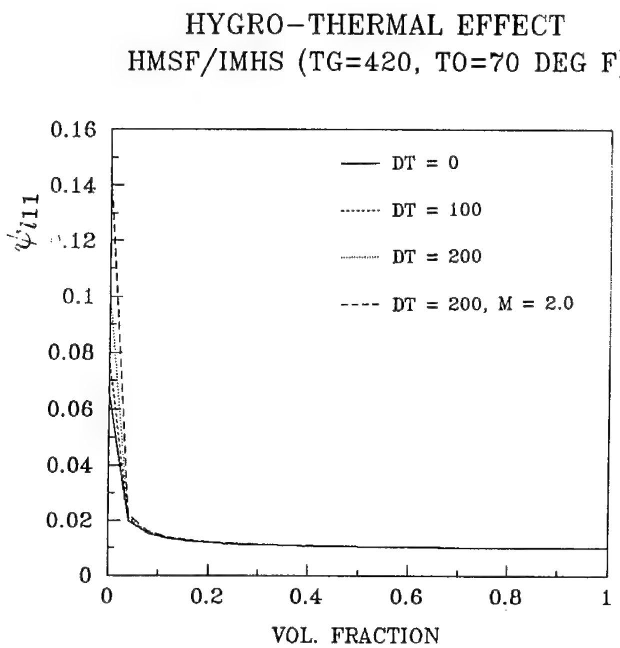
IN-PLANE SHEAR DAMPING

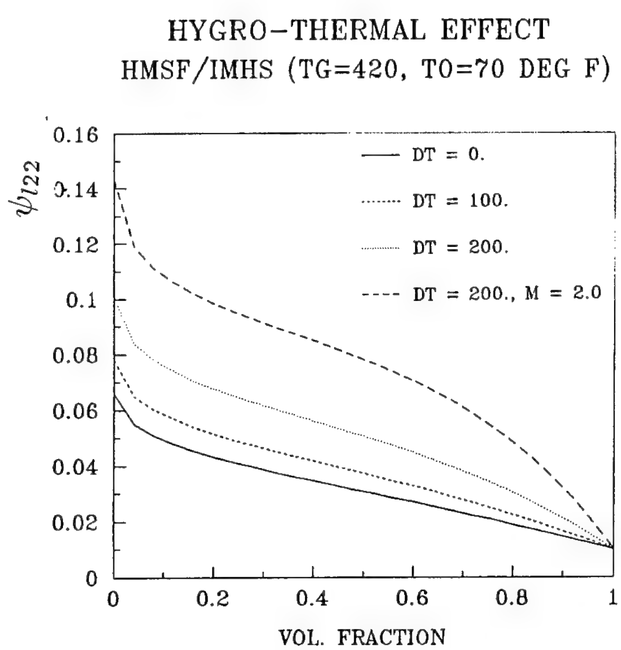


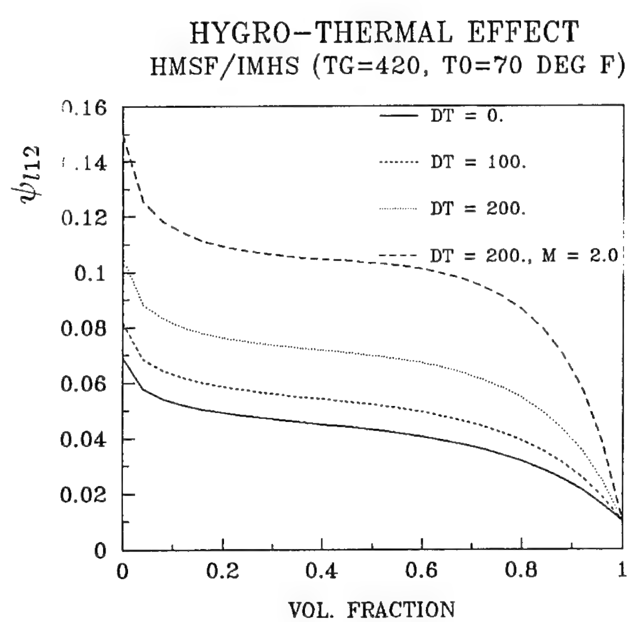
INTERPLY SHEAR DAMPING



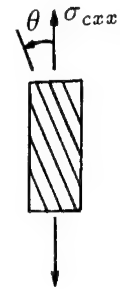

FRICTION EFFECT
HMSF/IMHS (0.005 IN)
SIGL11 = 10 KSI, MU = 0.1
ψ_{l11}/ψ_{mn}
0% BROKEN FIBERS
20% BROKEN FIBERS
50% BROKEN FIBERS
0
0.2
0.4
0.6
0.8
1
1.2
VOL. FRACTION


HYGRO-THERMAL EFFECT
HMSF/IMHS (TG=420, T0=70 DEG F)
ψ_{l11}
DT = 0
DT = 100
DT = 200
DT = 200, M = 2.0
0
0.02
0.04
0.06
0.08
0.1
0.12
0.14
0.16
0.2
0.4
0.6
0.8
1
VOL. FRACTION


HYGRO-THERMAL EFFECT
HMSF/IMHS (TG=420, T0=70 DEG F)
ψ_{l22}
DT = 0.
DT = 100.
DT = 200.
DT = 200., M = 2.0
0
0.02
0.04
0.06
0.08
0.1
0.12
0.14
0.16
0.2
0.4
0.6
0.8
1
VOL. FRACTION


HYGRO-THERMAL EFFECT
HMSF/IMHS (TG=420, T0=70 DEG F)
ψ_{l12}
DT = 0.
DT = 100.
DT = 200.
DT = 200., M = 2.0
0
0.02
0.04
0.06
0.08
0.1
0.12
0.14
0.16
0.2
0.4
0.6
0.8
1
VOL. FRACTION

OFF-AXIS DAMPING



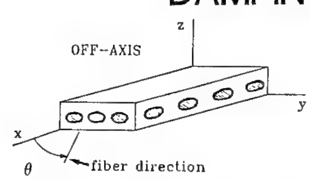
IN MATRIX FORM:

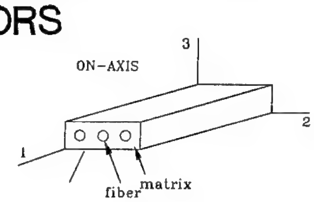
$$[\psi_c] = [R_\sigma]^T[\psi_l][R_\sigma^{-1}]^T$$

$$\psi_{cxx} = E_{cxx}[\frac{cos^2\theta}{E_{l11}}(cos^2\theta - \nu_{l12}sin^2\theta)\psi_{l11}+$$
$$+\frac{sin^2\theta}{E_{l22}}(sin^2\theta - \nu_{l21}cos^2\theta)\psi_{l22} + \frac{sin^2 2\theta}{4G_{l12}}\psi_{l12}]$$

- OFF-AXIS PLIES RESTRICT MOTION THROUGH COMBINED MODE DAMPING

MATRIX EQUATIONS USED IN OFF-AXIS DAMPING FACTORS



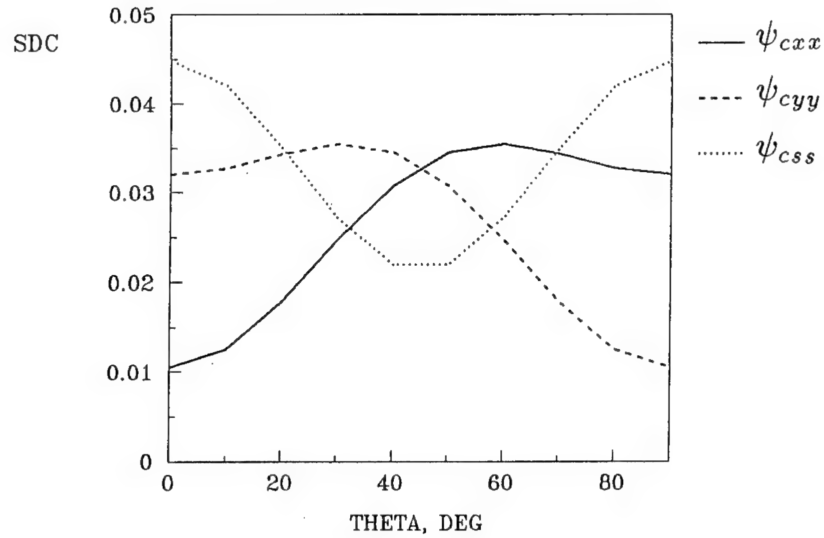


$$[\psi_c] = \begin{bmatrix} \psi_{cxx} & \psi_{cxy} & \psi_{cxs} \\ \psi_{cyx} & \psi_{cyy} & \psi_{cys} \\ \psi_{csx} & \psi_{csy} & \psi_{css} \end{bmatrix} \qquad [\psi_l] = \begin{bmatrix} \psi_{l11} & 0 & 0 \\ 0 & \psi_{l22} & 0 \\ 0 & 0 & \psi_{l12} \end{bmatrix}$$

$$[E_c]^{-1} = \begin{bmatrix} \frac{1}{E_{cxx}} & -\frac{\nu_{cyx}}{E_{cyy}} & \frac{\nu_{csx}}{G_{cxy}} \\ -\frac{\nu_{cxy}}{E_{cxx}} & \frac{1}{E_{cyy}} & \frac{\nu_{csy}}{G_{cxy}} \\ \frac{\nu_{cxs}}{E_{cxx}} & \frac{\nu_{cys}}{E_{cyy}} & \frac{1}{G_{cxy}} \end{bmatrix} \qquad [E_l]^{-1} = \begin{bmatrix} \frac{1}{E_{l11}} & -\frac{\nu_{l21}}{E_{l22}} & 0 \\ -\frac{\nu_{l21}}{E_{l11}} & \frac{1}{E_{l22}} & 0 \\ 0 & 0 & \frac{1}{G_{l12}} \end{bmatrix}$$

$$[E_c]^{-1} = [R_\sigma]^T[E_l]^{-1}[R_\sigma] \qquad [R_\sigma] = \begin{bmatrix} cos^2\theta & sin^2\theta & 2cos\theta sin\theta \\ sin^2\theta & cos^2\theta & -2cos\theta sin\theta \\ -cos\theta sin\theta & cos\theta sin\theta & cos^2\theta - sin^2\theta \end{bmatrix}$$

OFF-AXIS PLY DAMPING
50% HMSF/IMHS



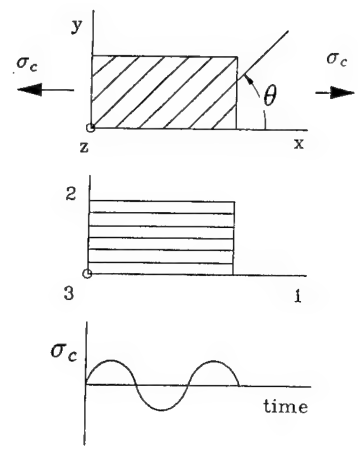
TEMPERATURE RISE DUE TO DAMPING
(INPLANE LOADING)

OFF-AXIS:

$$\delta T_c = \frac{c_F \omega t_c^2}{4k_{czz}} \{\sigma_c\}^T [\psi_c][E_c]^{-1}\{\sigma_c\}$$

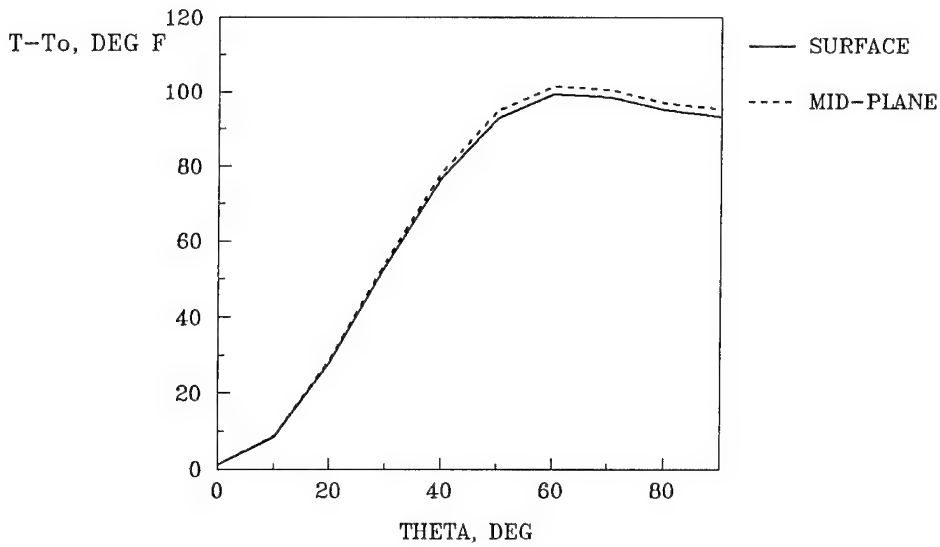
ALONG AXIS (UNIDIRECTIONAL):

$$\delta T_c = \frac{c_F \omega t_c^2}{4k_{c33}} \frac{\sigma_{caa}^2}{E_{caa}} \psi_{caa}$$



a DENOTES DIRECTION 1, 2, OR 12
δT_c = TEMPERATURE RISE (DEG. F)
c_F = CONVERSION FACTOR
ω = EXCITATION FREQUENCY
t_c = COMPOSITE THICKNESS
k_{czz} = k_{c33} = THERMAL CONDUCTIVITY
σ_c = STRESS
E_c = MODULUS
ψ_c = COMPOSITE SDC

TEMPERATURE RISE
CYCLIC STRESS SIGX=10 KSI AT 50 HZ
0.04 IN, 50% HT-S/IMHS



NEAR-FUTURE DIRECTIONS:

IN PRIORITY ORDER:

- LAMINATE THEORY FOR DAMPING
- FINITE ELEMENT DAMPING MATRICES
- MODAL DAMPING FOR COMPOSITE STRUCTURAL DYNAMICS ANALYSIS
- ISSUE ICAN WITH DAMPING
- SPECIALTY METHODS/COMPUTER CODES FOR SIZING COMPOSITE STRUCTURES WITH TAILORED DAMPING

SUMMARY OF RESULTS:

- o MICROMECHANICS FOR FIBER COMPOSITE PLY DAMPING AND TEMPERATURE RISE HAVE BEEN DEVELOPED
- o IT INCLUDES SEVEN DIFFERENT MODES OF DAMPING: 3 - NORMAL, 3 - SHEAR, AND 1 - INTERFACIAL
- o MECHANICS FOR OFF-AXIS DAMPING HAVE ALSO BEEN DEVELOPED
- o PREDICTIONS ARE IN GOOD AGREEMENT WITH LIMITED TEST DATA
- o HYGROTHERMAL ENVIRONMENTS HAVE SIGNIFICANT INFLUENCE TO PLY DAMPING
- o FIBER VOLUME RATIO WITHIN COMMERCIAL COMPOSITE RANGES HAS RELATIVELY SMALL SIGNIFICANCE ON LONGITUDINAL PLY DAMPING

SUMMARY OF RESULTS (CONT'D)

- o COMPOSITES WITH LOW THERMAL CONDUCTIVITIES EXHIBIT SUBSTANTIAL TEMPERATURE RISES UNDER CYCLIC LOADING
- o EXPERIMENTALLY MEASURED DAMPING WILL MOST LIKELY INCLUDE COMBINED MODES OF DAMPING AND CARE IS NEEDED TO INTERPRET THE RESULTS
- o THE MICROMECHANICS ARE USEFUL FOR:
    - o INTERPRETING EXPERIMENTAL DATA
    - o PLANNING EXPERIMENTS TO MEASURE IT
    - o ESTIMATING TEMPERATURE RISES
    - o LAMINATE THEORY OF DAMPING
    - o SIZING FIBER COMPOSITE STRUCTURES FOR TAILORED DAMPING

STRAIN ENERGY RELEASE RATE ANALYSIS OF DELAMINATION IN A TAPERED LAMINATE SUBJECTED TO TENSION 'OAD

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ABSTRACT

A tapered composite laminate subjected to tension load was analyzed using the finite-element method. The $([0_7/(\pm 45)]/\uparrow[(\pm 45)_3]/[0/(\pm 45)/0])_s$ glass/epoxy laminate has a $[\pm 45]_3$ group of plies dropped in three distinct steps, each 20 ply-thicknesses apart, thus forming a taper angle of 5.71 degrees. Steep gradients of interlaminar normal and shear stress on a potential delamination interface suggest the existence of stress singularities at the points of material and geometric discontinuities created by the internal plydrops. The delamination of the tapered laminate was assumed to initiate at the bottom of the taper on the -45/+45 interface indicated by the arrow in the laminate layup, and the delamination growth was simulated along the taper and into the thin region. The total strain-energy-release rate, G, and the Mode I and Mode II components of G, were computed at the delamination tip using the Virtual Crack Closure Technique. In addition, G was calculated from a global energy balance method. The strain-energy-release rate for a delamination growing in the thin laminate consisted predominantly of a Mode I (opening) component. For a delamination growing along the tapered region, the strain-energy-release rate was initially all Mode I but decreased with increasing delamination size until eventually it was all Mode II. These results indicated that a delamination initiating at the end of the taper will grow unstably along the taper and the thin laminate.

REFERENCES

[1] O'Brien, T. K.: Towards a Damage Tolerance Philosophy for Composite Materials and Structures. Presented at the 9th ASTM Symposium on Composite Materials: Testing and Design, Reno, Nevada, April 27-29, 1988. Also published as NASA TM-100548, May 1988.

[2] Rybicki, E. F.; and Kanninen, M. F.: A Finite-Element Calculation of Stress-Intensity Factors by a Modified Crack-Closure Integral. Engineering Fracture Mechanics, Vol. 9, pp. 931-938, 1977.

[3] Raju, I. S.: Calculation of Strain-Energy-Release Rates With Higher Order and Singular Finite Elements. Engineering Fracture Mechanics, Vol. 28, pp. 251-274, 1987.

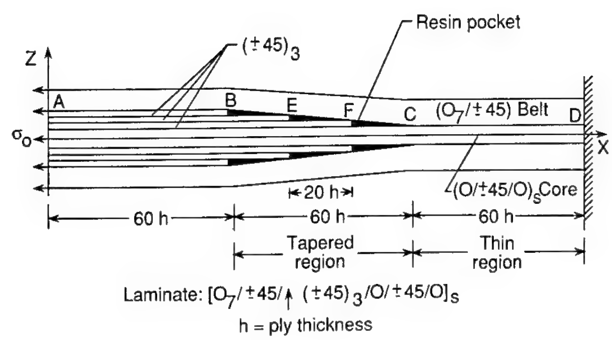

Fig. 1: Tapered laminate configuration and loading

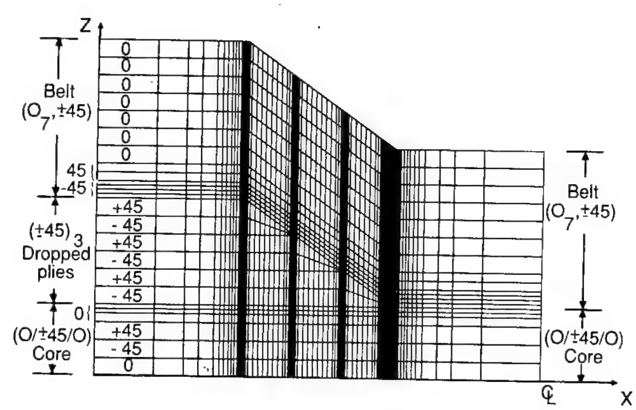

Fig. 2: Finite-element model of the tapered laminate

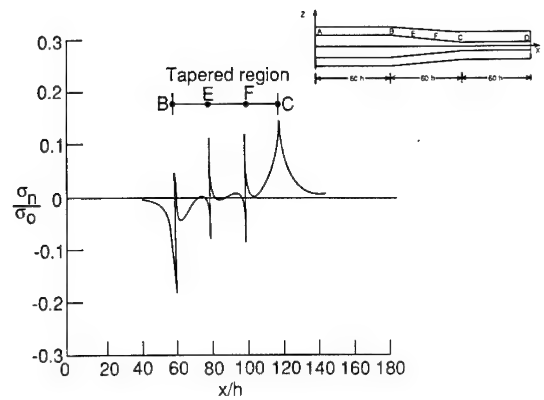

Fig. 3: Normalized interlaminar normal stress distribution along the tapered interface BEFC

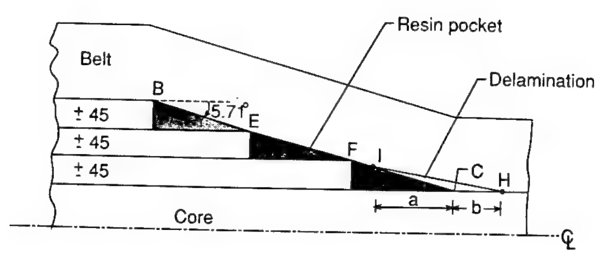

Fig. 4: Typical delamination

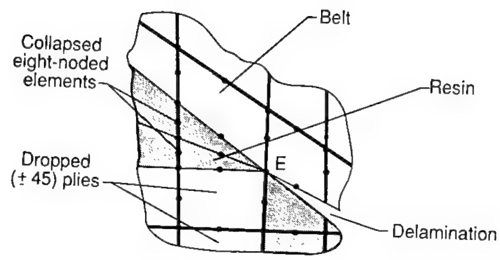

Fig. 5: Mesh detail showing delamination tip at point E

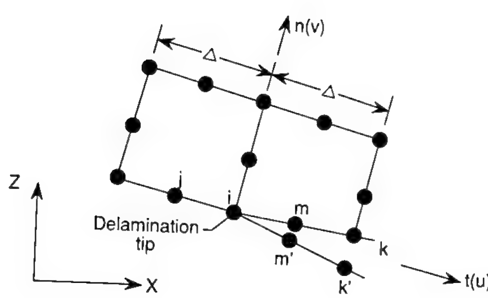

Fig. 6: Finite-element idealization near the delamination tip for calculating G using VCCT

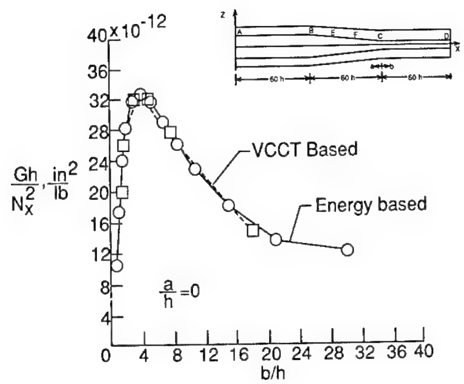

Fig. 7: Normalized total strain-energy-release rate at delamination tip H along interface CD

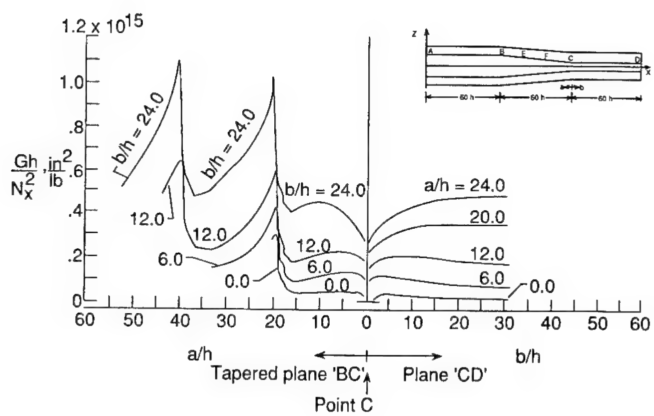

Fig. 8: Normalized total strain-energy-release rate at delamination tip I along interface CB and tip H along interface CD

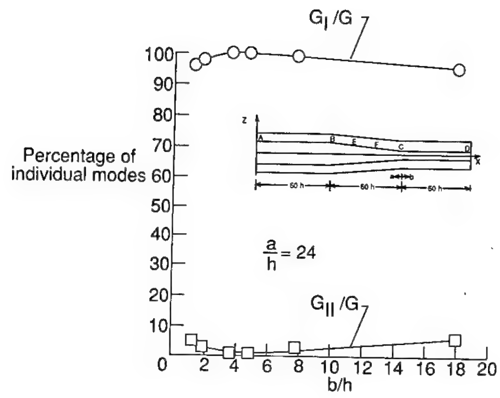

Fig. 9: G_I/G and G_{II}/G at delamination tip H along interface CD

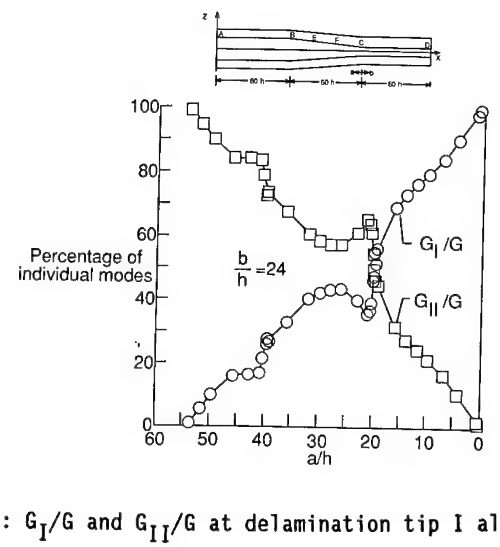

Fig. 10: G_I/G and G_{II}/G at delamination tip I along interface BC

A CONSISTENT SHEAR DEFORMATION THEORY FOR LAMINATED COMPOSITE PLATES

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ABSTRACT

To allow for shear deformation in laminated composite plates, the usual procedure involves introduction of a shear correction factor which, in general, is difficult to determine.

Recently [1], Hong derived constitutive equations for shear force resultants in each layer of a laminated plate from a mixed variational formulation of linear elastostatics. This variational formulation may be regarded as an extension, to laminated plate, of Reissner's original work for a homogeneous plate. Assuming that u_3, the transverse displacement, is independent of x_3 and u_α, the inplane displacements, are continuous and linear over the thickness of each layer, Hong showed that the shearing force in any lamina depends upon the shear deformation of all the laminae. Symbolically, the shearing forces over the kth layer are:

$$Q_\alpha^k = \sum_{j=1}^{N} \lambda_{\alpha\beta}^{kj} \; (\phi_\beta^j + u_{3,\beta}^j)$$

where N is the total number of layers, and ϕ_α^j the slope of u_α over the jth layer. The type and extent of the coupling, i.e. values of coefficients $\lambda_{\alpha\beta}^k$, depends upon the material properties of the laminae, their stacking sequence and the applied surface tractions. The theory is fairly general and many existing theories are seen to arise as specializations/approximations to the general theory.

In this paper, highlights of the theoretical approach as well as application of Hong's consistent shear deformation theory to vibration of a square laminated plate are presented. The implementation of the theory in a finite element program used Hughes' [2] "Heterosis" element. One quadrant of the square plate was discretized into 4, 16 and 36 finite elements. The results show good convergence characteristics and better approximation of the fundamental frequency of vibration as compared to the earlier approaches. The influence of coupling in shear resultants in the laminate appears to be more significant for higher frequences. The coupled shear deformation theory is theoretically consistent and easy to implement in a computer program.

Further work on evaluation of stresses in the plate, study of delamination and application of the theory to transient dynamic response is in progress and will be reported in due course.

ACKNOWLEDGEMENTS

The research reported is part of the work done at The Ohio State University under ASD/AFWAL Grant F33625-85-C3213. Dr. G. P. Sendeckyj is the program manager. Assistance provided by The Ohio State University Instruction and Research Computer Center and the Ohio Supercomputer Center is gratefully acknowledged.

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1. S. J. Hong, A Consistent Shear Deformable Theory for the Vibration of Laminated Plates, Ph.D. Dissertation, The Ohio State University, Columbus, Ohio, 1988.

2. T. J. R. Hughes and M. Cohen, "The 'Heterosis' Finite Element for Plate Bending," Computers and Structures, Vol. 46, pp. 203-222, 1978.

A CONSISTENT SHEAR THEORY OF LAMINATED PLATES

by

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R. S. SANDHU

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THE OHIO STATE UNIVERSITY

OBJECTIVE

TO PROPERLY ALLOW FOR SHEAR DEFORMATION IN LAMINATED COMPOSITE PLATES

APPROACH

- DISCRETE LAMINATE THEORY
- COUPLED CONSTITUTIVE EQUATIONS FOR SHEAR

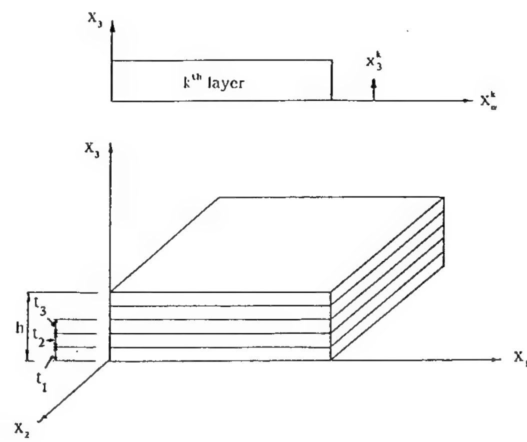

Figure 1: A LAMINATED PLATE

DISCRETE LAMINATE THEORY

- KINEMATICS

$$u_\alpha^k(x_i,t)=\bar{u}_\alpha(x_\beta,t)+x_3^k\,\phi_\alpha(x_\beta,t)$$

$$u_3^k(x_i,t)=w^k(x_\beta,t)$$

$$e_{\alpha\beta}^k=\bar{u}_{(\alpha,\beta)}+x_3^k\,\phi_{(\alpha,\beta)}$$

$$e_{\alpha 3}^k=\frac{1}{2}\left(\phi_\alpha^k+w_{,\alpha}^k\right)$$

$$e_{33}^k=0$$

- EQUILIBRIUM

$$N_{\alpha\beta,\beta}^k+\left(T_\alpha^k-T_\alpha^{k-1}\right)+F_\alpha^k-P^k\ddot{\bar{u}}_\alpha^k-R^k\ddot{\phi}_\alpha^k=0$$

$$M_{\alpha\beta,\beta}^k-Q_\alpha^k+G_\alpha^k+t_kT_\alpha^k-R^k\ddot{\bar{u}}_\alpha^k-I^k\ddot{\phi}_\alpha^k=0$$

$$Q_{\alpha,\alpha}^k+\left(T_3^k-T_3^{k-1}\right)+F_3^k-P_3^k\ddot{w}^k=0$$

$$\left(F_\alpha^k,G_\alpha^k\right)=\int_0^{t_k}\left(1,x_3^k\right)f_\alpha^k\,dx_3^k$$

$$F_3^k=\int_0^{t_k}f_3^k\,dx_3^k$$

$$\left(P^k,R^k,I^k\right)=\int_0^{t_k}\{1,x_3^k,(x_3^k)^2\}\,\rho^k\,dx_3^k$$

$$T_i^k=\sigma_{i3}^k\left(x_3^k=t_k\right)=\sigma_{i3}^{k+1}\left(x_3^{k+1}=0\right)$$

$$T_i^{k-1}=\sigma_{i3}^k\left(x_3^k=0\right)=\sigma_{i3}^{k-1}\left(x_3^{k-1}=t_{k-1}\right)$$

- CONTINUITY CONDITIONS

$$\bar{u}_\alpha^{k+1}=\bar{u}_\alpha^k+t_k\,\phi_\alpha^k$$

$$w^{k+1}=w^k$$

$$\sigma_{i3}^k\left(x_3^k=t_k\right)=\sigma_{i3}^{k+1}\left(x_3^{k+1}=0\right)$$

- CONSTITUTIVE RELATIONS

$$\begin{Bmatrix}N_{\gamma\delta}^k\\M_{\gamma\delta}^k\end{Bmatrix}=\begin{bmatrix}A_{\alpha\beta\gamma\delta}^k & B_{\alpha\beta\gamma\delta}^k\\B_{\alpha\beta\gamma\delta}^k & D_{\alpha\beta\gamma\delta}^k\end{bmatrix}\begin{Bmatrix}\bar{e}_{\alpha\beta}^k\\\kappa_{\alpha\beta}^k\end{Bmatrix}$$

$$\bar{e}_{\alpha\beta}^k=\bar{u}_{(\alpha,\beta)}^k$$

$$\kappa_{\alpha\beta}^k=\phi_{(\alpha,\beta)}^k$$

$$\left(A^k,B^k,D^k\right)=\left(t_k,\frac{t_k^2}{2},\frac{t_k^3}{3}\right)E^k$$

$$E_{\alpha\beta\gamma\delta}^k=E_{\alpha\beta\gamma\delta}^k-\frac{E_{\alpha\beta33}^k}{E_{3333}^k}\,E_{33\gamma\delta}^k$$

CONSTITUTIVE EQUATION FOR SHEAR

- PRELIMINARIES

$$\Omega = \int_R \Big\{ \frac{1}{2}\sigma_{\alpha\beta}u_{\alpha,\beta} + \sigma_{\alpha 3}(u_{\alpha,3}+u_{3,\alpha}) + \sigma_{33}u_{3,3} - \sigma_{\alpha 3}\bar{e}_{\alpha 3} - \frac{1}{2}\sigma_{33}\bar{e}_{33}\Big\} dR + \int_{S_2} u_i \hat{t}_i ds$$

$$\bar{e}_{i3} = C_{i3\gamma\delta}\sigma_{\gamma\delta} + 2C_{i3\gamma 3}\sigma_{\gamma 3} + C_{i333}\sigma_{33}$$

- CONSTITUTIVE EQUATION

$$\int_A \left(\sum_{k=1}^{N} \int_0^{t_k} \delta\sigma^k_{\alpha 3}\left(u^k_{\alpha,3}+u^k_{3,\alpha}-2\bar{e}^k_{\alpha 3}\right) dx^k_3 \right) dA = 0$$

where

$$\sigma^k_{\alpha 3} = \zeta^k_1 Q^k_\alpha + \zeta^k_2 T^{k-1}_\alpha + \zeta^k_3 T^k_\alpha \qquad \text{REISSNER}$$

$$\zeta^k_1 = 6\frac{x^k_3}{t^2_k}\left(1-\frac{x^k_3}{t_k}\right)$$

$$\zeta^k_2 = \left(1-\frac{x^k_3}{t_k}\right)\left(1-3\frac{x^k_3}{t_k}\right)$$

$$\zeta^k_3 = \frac{x^k_3}{t_k}\left(3\frac{x^k_3}{t_k}-2\right)$$

and

$$\bar{e}^k_{\alpha 3} = S^k_{\alpha\beta}\sigma^k_{\beta 3}$$

- VARIATION

$$\Delta\Omega = \int_R \{\sigma_{ij}\delta u_{i,j} + \delta\sigma_{\alpha 3}(u_{\alpha,3}+u_{3,\alpha}-2\bar{e}_{\alpha 3}) + \delta\sigma_{33}(u_{3,3}-\bar{e}_{33})\} dR + \int_{S_2} \delta u_i \hat{t}_i ds = 0$$

For a laminate

$$\Delta\Omega = \int_A \Big(\sum_{k=1}^{N} \int_0^{t_k} [\sigma^k_{ij}\delta u^k_{i,j} + \delta\sigma^k_{\alpha 3}\left(u^k_{\alpha,3}+u^k_{3,\alpha}-2\bar{e}^k_{\alpha 3}\right) + \delta\sigma^k_{33}\left(u_{3,3}-\bar{e}^k_{33}\right)] dx^k_3 \Big\} dA + \int_A \left(\delta u^+_i \hat{t}^+_i + \delta u^-_i \hat{t}^-_i\right) dA + \int_{S_2} \left(\sum_{k=1}^{N} \int_0^{t_k} \delta u^k_i \hat{t}^k_i dx^k_3 \right) ds = 0$$

SUN 1973
MURAKAMI 1986

Integration w. r. t. x_3 ⇒

$$\int_A \sum_{k=1}^{N} \left\{ \delta [H^k_\alpha]^T \left(\{L^k\}(\phi^k_\alpha + w^k_{,\alpha}) - [N^k][H^k_\beta S_{\beta\alpha}] \right) \right\} dA = 0$$

$$[H^k_\alpha]^T = [Q^k_\alpha, T^{k-1}_\alpha, T^k_\alpha]$$

$$[L^k] = [L^k_1, L^k_2, L^k_3] = [1, 0, 0]$$

$$[N^k] = \frac{1}{30t_k}\begin{pmatrix} 36 & -3t_k & -3t_k \\ & 4t^2_k & -t^2_k \\ \text{symm.} & & 4t^2_k \end{pmatrix}$$

No. of Equations 2(2N-1)

- COUPLED SHEAR THEORY

$$\left(\phi_\alpha^k + w_{,\alpha}^k\right) = \sum_{j=1}^{N} \mu_{\alpha\beta}^{kj} Q_\beta^j$$

$$Q_\alpha^k = \sum_{j=1}^{N} \lambda_{\alpha\beta}^{kj} \left(\phi_\beta^j + w_{,\beta}^j\right)$$

MINDLIN PLATE

$$Q_\alpha = \sum_{k=1}^{N} Q_\alpha^k = [\sum_{k=1}^{N} \sum_{j=1}^{N} \lambda_{\alpha\beta}^{kj}](\phi_\beta + w_{,\beta})$$

HOMOGENEOUS ISOTROPIC PLATE

$$Q_\alpha = \frac{5}{6} h \lambda_{\alpha\beta} (\phi_\beta + w_{,\beta}) + \frac{1}{12} h \left(T_\alpha^+ + T_\alpha^-\right)$$

TRACTION FREE SURFACES $\Longrightarrow$ REISSNER

AN APPLICATION

- VIBRATION OF A SANDWICH PLATE

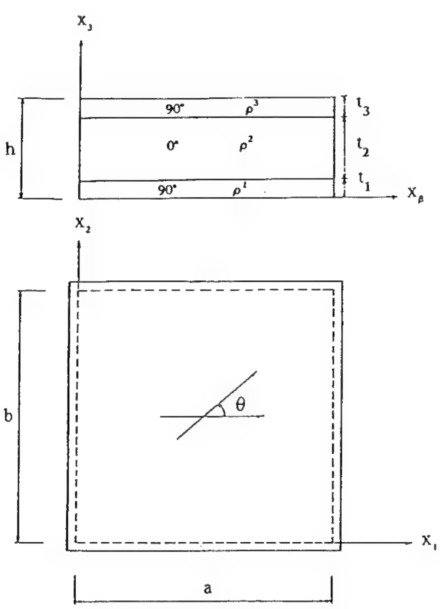

Figure 2: PLAN AND SECTION OF A LAMINATE

Case	t_1/h	t_2/h	t_3/h	ρ^1/ρ^2	ρ^3/ρ^2	$\bar{Q}_{66}^1/\bar{Q}_{66}^2$	$\bar{Q}_{66}^3/\bar{Q}_{66}^2$
I	0.1	0.8	0.1	1.0	1.0	1	1
II	0.1	0.8	0.1	1.0	1.0	10	10
III	0.1	0.8	0.1	1.0	1.0	50	50

- Ratios of orthotropic elastic constants:

$$\bar{Q}_{11}:\bar{Q}_{12}:\bar{Q}_{22}:\bar{Q}_{44}:\bar{Q}_{55}:\bar{Q}_{66} = 3.802:0.879:1.996:1.015:0.608:1.0$$

Table 1: MATERIAL PROPERTIES: THREE CASES

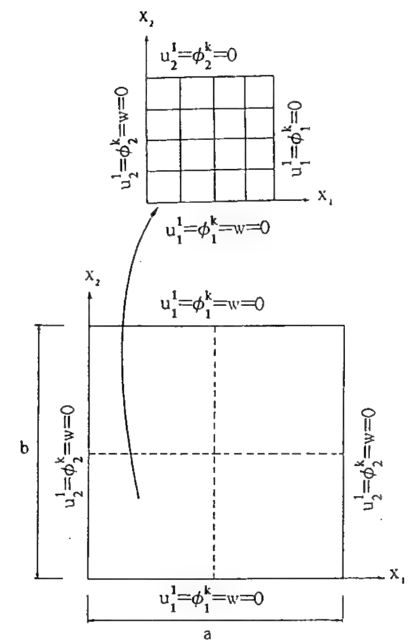

Figure 3: FINITE ELEMENT DISCRETIZATION

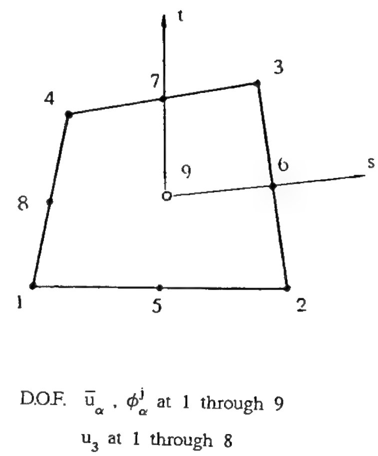

Figure 4: THE 'HETEROSIS' ELEMENT

Mesh	Case I	Case II	Case III
4	0.094697 (2.4%)	0.19682 (2.9%)	0.31097 (3.8%)
16	0.092952 (0.5%)	0.19472 (1.8%)	0.30919 (3.2%)
36	0.092900 (0.4%)	0.19463 (1.7%)	0.30911 (3.1%)
Exact 3-D (Srinivas)	0.09248	0.19132	0.29954

Table 2: NON-DIMENSIONALIZED FUNDAMENTAL FREQUENCY

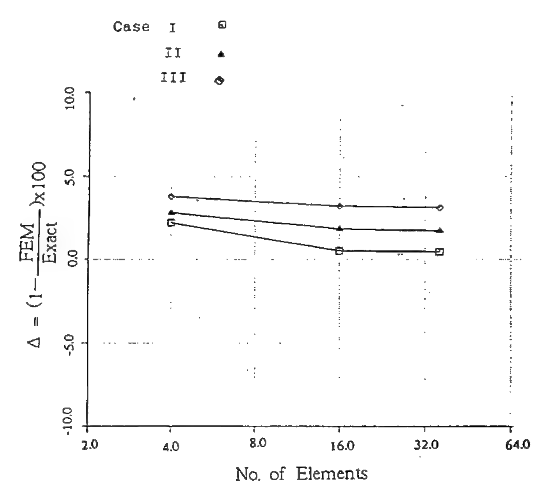

Figure 5: EFFECT OF MESH REFINEMENT

Mesh	Case I	Case II	Case III
4	0.09498 (2.7%)	0.19710 (3.0%)	0.31489 (5.1%)
16	0.09317 (0.7%)	0.19584 (2.3%)	0.31305 (4.5%)
36	0.09341 (0.7%)	0.19805 (3.5%)	0.32232 (7.6%)
Exact (Srinivas)	0.09248	0.19132	0.29954

Table 3: EFFECT OF MESH REFINEMENT K=5/6

Non-Dimensionalized Fundamental Frequency			
		Present Sol.	
Exact Sol. (Srinivas)	Reddy (4x4 mesh)	2x2 mesh	4x4 mesh
0.09315	0.0963 (3.3%)	0.0951 (2.2%)	0.09319 (0.04%)

* Value in parentheses indicates percentage error.

Table 4: ISOTROPIC LAMINATE: COMPARISON WITH REDDY'S SOLUTION

CONCLUSIONS

- COUPLED SHEAR THEORY
  1. IS MORE GENERAL
  2. GIVES BETTER FINITE ELEMENT APPROXIMATION TO FREQUENCIES
  3. IS NO MORE DIFFICULT TO IMPLEMENT THAN OTHER DISCRETE LAMINATE THEORIES
  4. GIVES ESTIMATED HIGHER FREQUENCIES CONSIDERABLY LOWER THAN THE UNCOUPLED SHEAR THEORY

COMPRESSION RESPONSE OF THICK-SECTION COMPOSITES

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ABSTRACT

The compressive response of fiber-reinforced composite materials has been of considerable interest to materials scientists and engineers since these materials were first considered for structural applications. This interest has resulted in numerous research programs addressing the theoretical and experimental response of composites to compressive loading [1]. These research programs have dealt exclusively with the response of composite materials 0.25 inches in thickness and less. As composite materials become more attractive for use in large Navy structures, the need to understand the mechanical response of composites greater than 0.25 inches in thickness becomes a necessity.

The objective of this program is to investigate the compressive response of composite materials greater than 0.25 inches thick to assess the design advantages/limitations of composites for large, thick Navy structures. The approach taken to investigate these issues is to design and refine a compression test fixture that allows the testing of composites up to one inch in thickness and greater, to evaluate the effects of constituents, fiber orientation, and thickness on compressive response, and determine if the failure mechanisms observed for thick composites are adequately accounted for in existing failure theories that have been developed for observed failure mechanisms in thin composite materials.

This program is currently ongoing and progress to date includes fabrication of 0.25 inch and 0.50 inch thick samples with carbon and S2 glass reinforcements, fabrication and evaluation of a DTRC thick compression test fixture for 0.25 and 0.50 inch samples, and failure data on the 0.25 inch specimens. The 0.50 inch specimens allow the determination of NU13 and are instrumented to provide this data.

Inherent lamina waviness has been observed in the $[0_2, 90]_s$ specimens evaluated in this program and this waviness has been quantified to theoretically determine the effect of fiber curvature and misalignment on local stress state.

REFERENCES

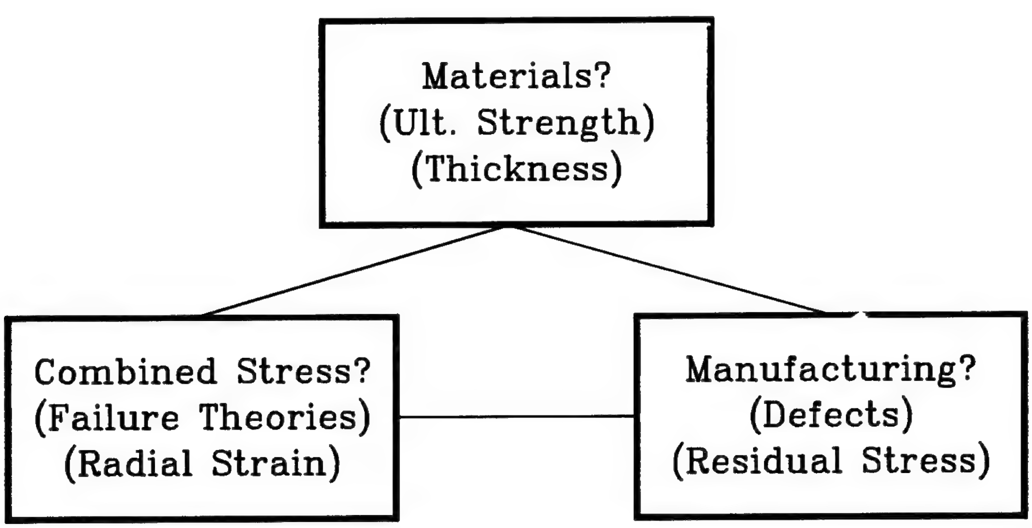
1. Camponeschi, E. T., Jr., "Compression of Composite Materials: A Review," DTRC-87/050, November 1987.

OBJECTIVE

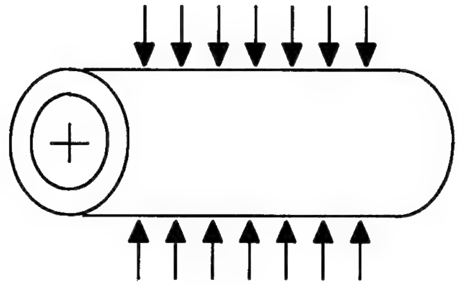
DEVELOP AN UNDERSTANDING OF COMPRESSION FAILURE
FOR THICK–SECTION COMPOSITES

COMPOSITE COMPRESSION PERFORMANCE

	CARBON	vs.	GLASS
COUPON	210 ksi → 140 ksi		180 ksi → 120 ksi
CYLINDER	80 ksi?		120 ksi



TECHNICAL ISSUES



TRIAXIAL STRESS
RESIDUAL STRESS
MANUFACTURING FLAWS
STRENGTH/BUCKLING

CONSTITUENT EFFECTS
ORIENTATION EFFECTS
THICKNESS EFFECTS
EFFECT OF FLAWS
STRENGTH/BUCKLING

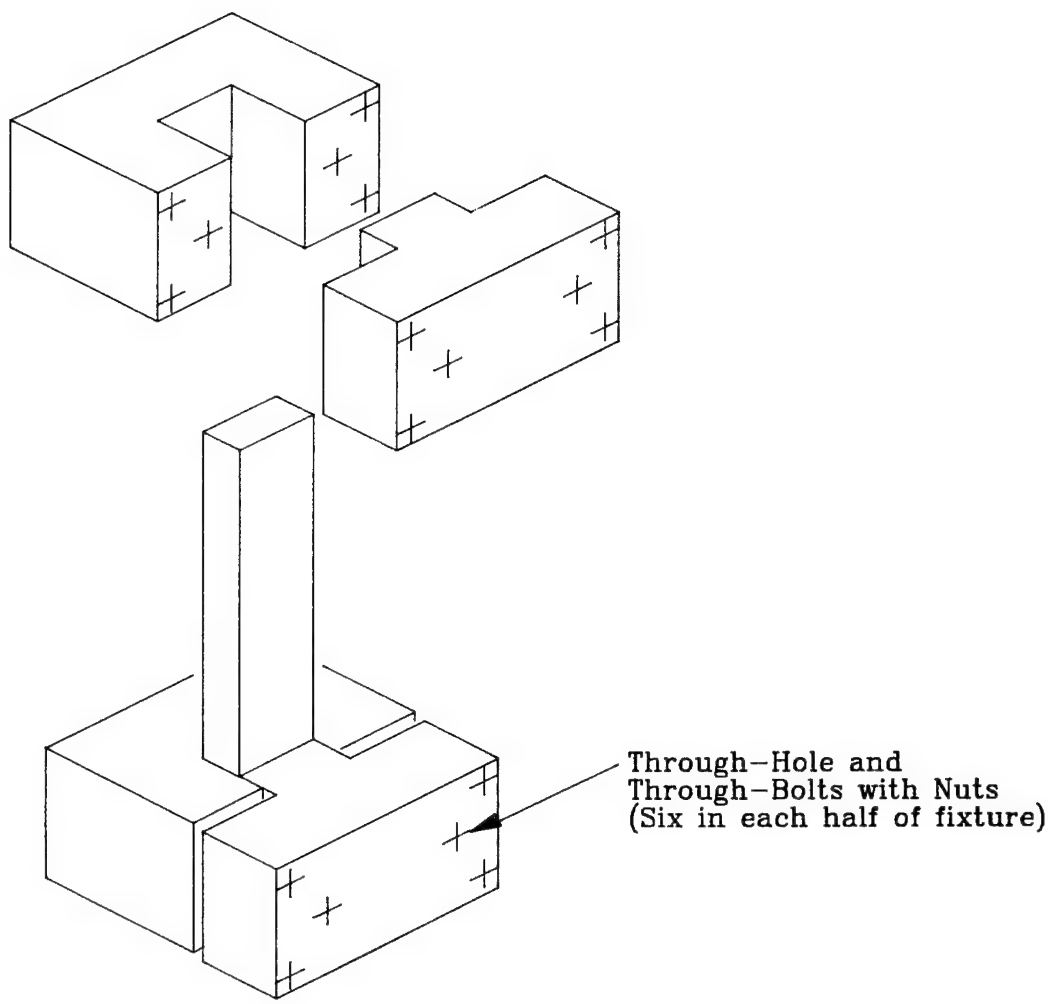
FAILURE BASED DESIGN

TECHNICAL APPROACH

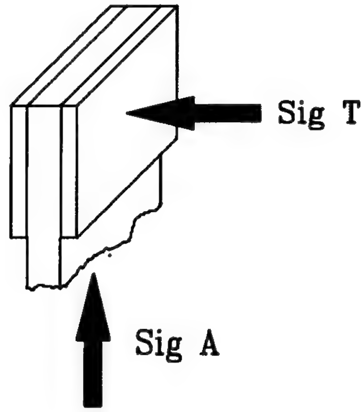
- THICK–SECTION FAILURE THEORY BASED ON OBSERVED FAILURE MODES
- DEVELOP THICK–SECTION TEST
- GRAPHITE, GLASS
- EPOXY, THERMOPLASTIC
- (0), (0,90), OFF–AXIS
- 0.25–1.0 INCHES THICK
- CONDUCT FAILURE ANALYSES

TEST METHOD REQUIREMENTS

END–LOADED, UNSUPPORTED GAGE LENGTH
FRICTIONLESS
NO LOAD ECCENTRICITIES
VARIABLE SPECIMEN THICKNESS
VARIABLE SPECIMEN WIDTH
VARIABLE SPECIMEN LENGTH/THICKNESS
ALLOWS THICK–SECTION TESTING
ALLOWS FURTHER SCALE–UP



THROUGH–THICKNESS STRESS

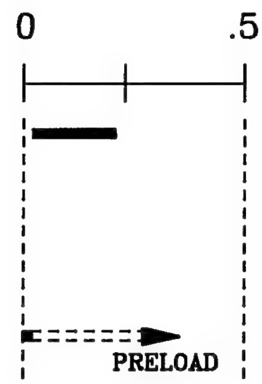


IITRI $\quad .029 < \dfrac{\text{Sig T}}{\text{Sig A}} < .224$

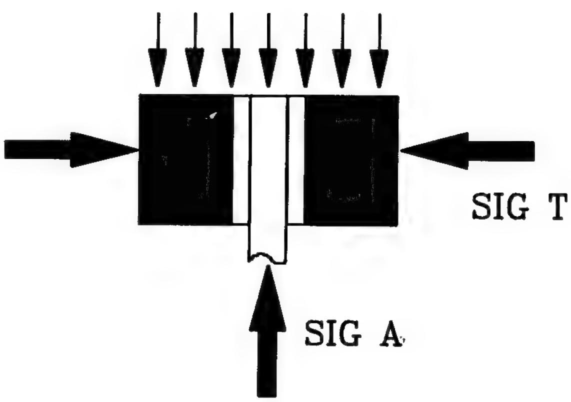
DTRC

Carbon $\quad \dfrac{\text{Sig T}}{\text{Sig A}} = .007$

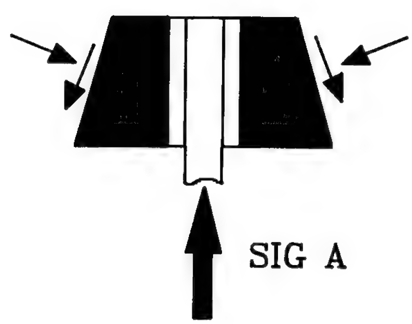
Glass $\quad \dfrac{\text{Sig T}}{\text{Sig A}} = .023$



DTRC



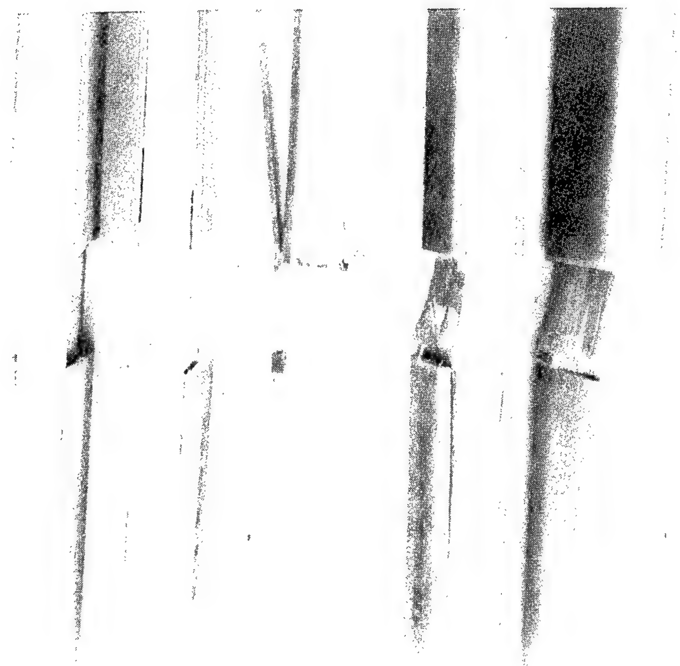
IITRI

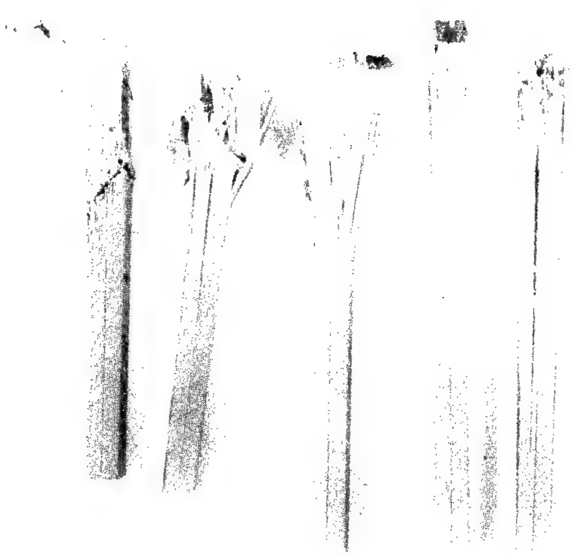


$$\frac{\text{Sig T}}{\text{Sig A}} = \frac{NU_{13}\, E_b\, E_3\, A_T\, t}{E_1 [\, L_b\, E_3\, A_s + E_b\, A_T\, t \,]}$$

$$\frac{\text{Sig T}}{\text{Sig A}} = \frac{t_{spec}}{L_{tabs}} \frac{[\cos(\Theta) - u \sin(\Theta)]}{[u \cos(\Theta) + \sin(\Theta)]}$$

After Bogetti





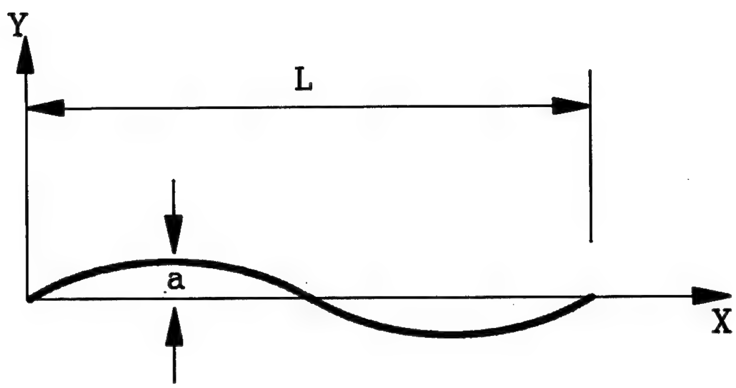
COMPRESSION STRENGTH
48 PLY HYE 9137B S2 GLASS/EPOXY

Orien.	L/T	No. Spec.	Strength (ksi) (S.D.)	Stiffness (Msi) (S.D.)	Fiber Vol. Fract.	Failure
[0]	3:1	5	165.0 (17.50)	--	55.4%	4 Gage 1 Grip
[0]	5:1	5	184.8 (18.00)	--	55.4%	3 Gage 2 Grip
$[0_2/90]$	3:1	7	150.4 (12.1)	0.694 (.017)	56.2%	3 Gage 4 Grip
$[0_2/90]$	5:1	6	144.7 (7.05)	0.717 (.038)	56.2%	2 Gage 4 Grip

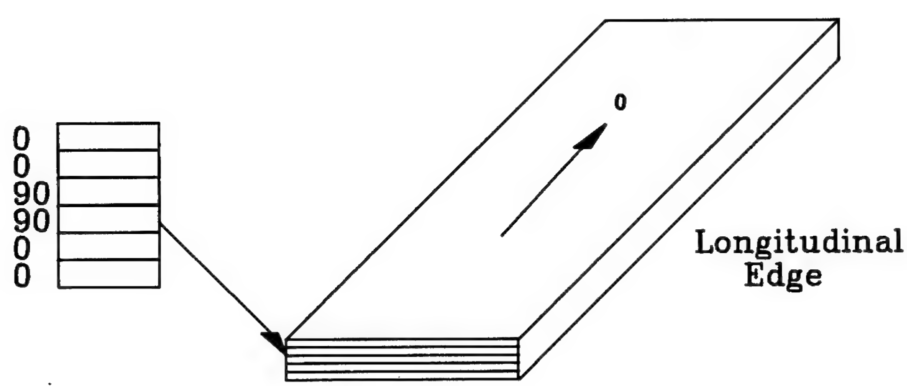
COMPRESSION STRENGTH
48 PLY AS4/3501-6 CARBON/EPOXY

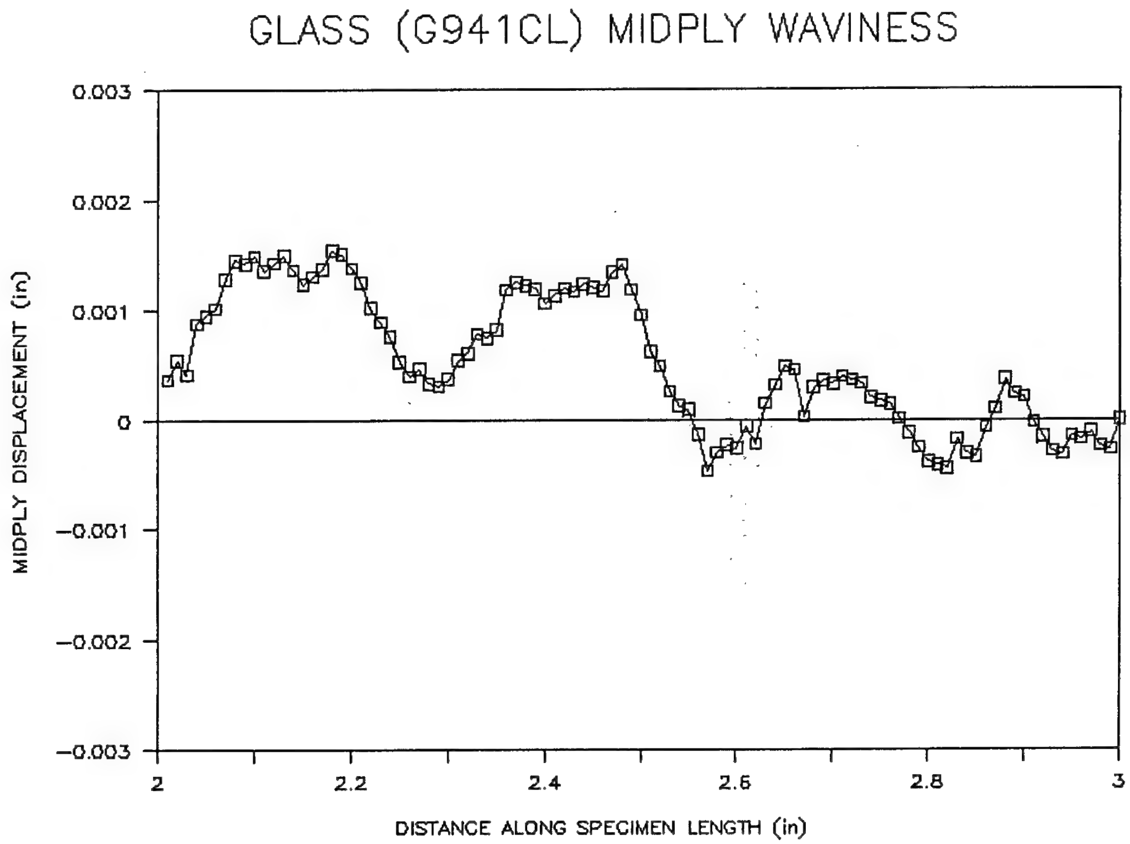
Orien.	L/T	No. Spec.	Strength (ksi) (S.D.)	Stiffness (Msi) (S.D.)	Fiber Vol. Fract.	Failure
[0]	3:1	2	136.4 (19.87)	--	60.0%	2 Grip
[0]	5:1	3	174.8 (13.85)	--	60.0%	3 Grip
$[0_2/90]$	3:1	7	130.0 (19.0)	12.2 (.465)	62.8%	4 Gage 3 Grip
$[0_2/90]$	5:1	8	130.7 (9.69)	12.6 (.180)	62.8%	5 Gage 3 Grip

LAMINA WAVINESS GEOMETRY AND ORIENTATION

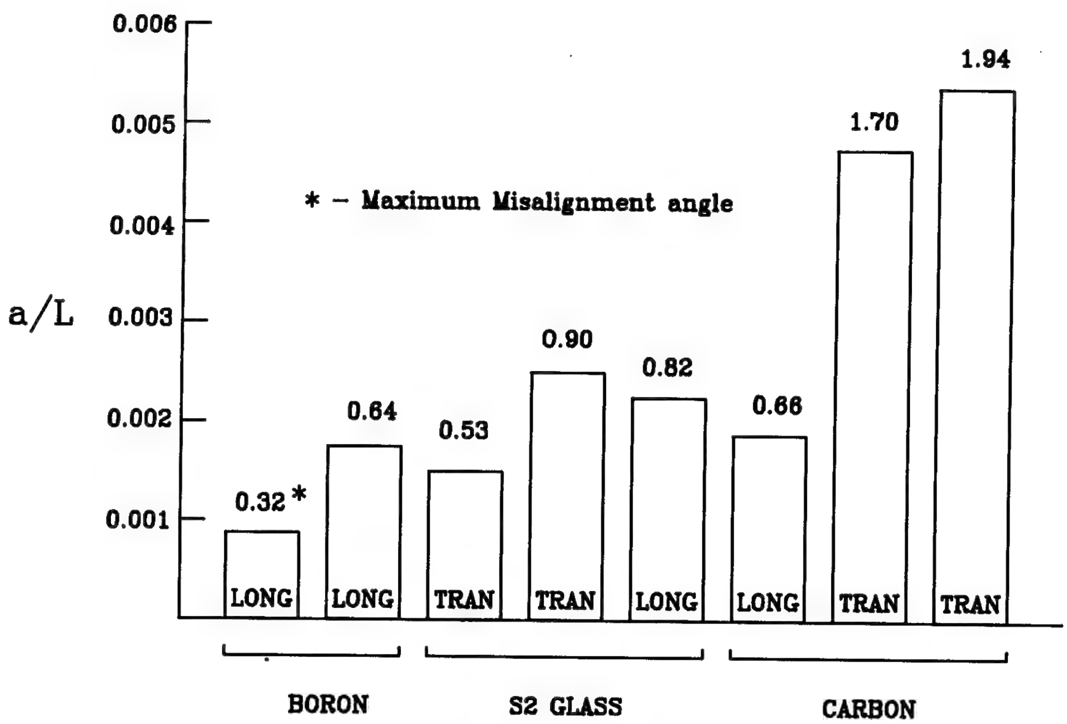


$$Y = a \sin\left[\frac{2\,(Pi)\,X}{L}\right]$$





MIDPLANE WAVINESS CHARACTERISTICS



CONCLUSIONS

- COMPRESSION RESPONSE IS A FUNCTION OF SAMPLE GEOMETRY AND TEST CONFIGURATION
- SHEAR LOAD INTRODUCTION PROVIDES GREATER RESTRAINT IN GRIPPED REGION THAN END-LOADING
- LAMINA WAVINESS IS A FUNCTION OF FIBER STACKING SEQUENCE FOR CARBON COMPOSITES
- CARBON REINFORCED SAMPLES MORE SENSITIVE TO LOAD INTRODUCTION THAN FIBERGLASS REINFORCED SAMPLES
- UNIDIRECTIONAL STRENGTH DETERMINED FROM BIDIRECTIONAL LAMINATES IS GREATER THAN DETERMINED FOR UNIDIRECTIONAL LAMINATES

DYNAMIC RESPONSE OF COMPOSITE MATERIALS FOR MARINE STRUCTURAL APPLICATIONS

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ABSTRACT

The use of composite materials for marine structural applications has been an area of sporadic research for the past 25 years [1,2]. The advantages of composite materials over current materials are substantial in several areas. Structurally, composites have high strength to density ratios making them attractive as materials for applications in deep diving submersibles. It is clear that despite the potentially large payoff involved in using composites there remain a number of issues that must be addressed before these materials can be applied in widespread undersea structural areas. One of these issues is the dynamic response of a composite structure when subjected to severe, rapidly applied pressure loading. Understanding the response of the material and structure to this loading condition is of primary importance in determining and increasing the durability of the composite structure.

The goal of this program is to develop the experimental and analytical techniques required to assess the dynamic capabilities of proposed composite structural and material concepts. Currently these techniques are available for metallic materials but the direct applicability to composite materials has not been demonstrated.

The approach to addressing this problem has been divided into two parts in this program. One part is focused on characterizing the composite materials of interest under the prescribed loading conditions. This portion of the program will seek to develop thick section, dynamic material specimen tests to investigate the effect of high rate loading on various composite material systems. The second part of the program will investigate the dynamic response of generic composite structures through use of small scale model tests and finite element analysis techniques. It is the structural portion of this task that will be addressed in this paper.

The focus of the structural portion of the program thus far has been the testing and analysis of a small scale graphite/epoxy model. The 8 in. diameter model is dynamically tested at shallow submergence depth in a test pond at the David Taylor Research Center. The model, an unstiffened thick-walled graphite/epoxy cylinder, was fabricated by Hitco Inc. for a previous composite structures research program. Celion 6K fibers and E707 epoxy matrix were placed in a 2 to 1 orientation (two plies in the circumferential direction, one ply in the longitudinal direction). The model originally had a wall thickness of 0.6 in. The model was modified for this program by reducing the wall thickness in the center portion but retaining the full thickness at the ends in order to reduce the possibility of a shear failure at the end closure. The dimensions of the model are; 7 in. length, 8.24 in. outside diameter, and 7.64 in. inside diameter in the test section. Aluminum flat plate end closures are used to seal the ends of the cylinder.

An explosive detonated underwater develops two primary types of pressure loadings. These are an initial rapidly-rising, exponentially decaying shock wave and a later slower-rising, longer duration bubble pulse. The intent of these tests was to apply only the relatively well characterized shock wave loading to the model. For this reason, the explosive charge was placed in close proximity to the water surface to allow the bubble to vent to the atmosphere during its initial expansion thus eliminating the bubble pulse. Four tests were conducted on the model with successively increasing peak pressure loadings. Additional tests will be performed if necessary to produce significant damage. The model was non-destructively evaluated using C-scan ultrasonic inspection after all but the first test. The model was instrumented with biaxial strain gages at five locations and ten channels of data were recorded during the test. The measured strains were compared with the analytical predictions to evaluate the accuracy of the finite element analytical model.

The finite element method was used to analytically model the response of the composite cylinder. The ADINA-S finite element program [3], which includes fluid-structure interaction, was used for this analysis. The analytical model consisted of first 83

three-dimensional elements and then was refined further to include 141 elements. Generalized orthotropic material properties were used to simulate the composite material of the actual structure. For a $[90_2/0]$ layup such as this one, orthotropic properties accurately represent the global material response, however, the lamina level stresses are not determined. The properties used are static values but these can be changed at a later date to incorporate the effect of the high loading rates. Initially, the analytical effort is geared to reproducing the general response of the model. The accuracy is determined by comparing the analytical and experimental strain histories. Comparisons made for the four tests conducted thus far are presented and show good correlation when one considers the complexity of the loading and the uncertainty in the material properties. Once failure mechanisms have been identified, the model can be refined to specifically investigate those failure modes. The goal of the refined model is to begin developing a predictive capability for the failure modes of the composite structures subjected to this loading condition.

REFERENCES

1. Hom, K. and W. P. Couch, "Investigation of Filament Reinforced Plastics for Deep Submergence Application," David Taylor Model Basin Report 1824 (Nov 1966).

2. Garala, H. J., "Experimental Evaluation of Graphite/Epoxy Composite Cylinders Subjected to External Hydrostatic Compressive Loading," in: Proceedings of the 1987 Spring Conference on Experimental Mechanics, 14-19 June 1987, Houston, Texas, Society for Experimental Mechanics (SEM), pp. 948-951, Bethal CT (1987).

3. ADINA-S is a general-purpose computer program developed by K. J. Bathe at Massachusetts Institute of Technology, Cambridge MA (1978) and modified by DTRC Code 175.

DYNAMIC RESPONSE OF COMPOSITE MATERIALS FOR MARINE STRUCTURAL APPLICATIONS

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DAVID TAYLOR RESEARCH CENTER

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OBJECTIVES

- DEVELOP EXPERIMENTAL TECHNIQUES REQUIRED TO ASSESS THE DYNAMIC CAPABILITIES OF THICK SECTIONED COMPOSITE MATERIALS AND STRUCTURAL CONCEPTS.
- DEVELOP ANALYTICAL TECHNIQUES REQUIRED TO ASSESS THE DYNAMIC RESPONSE OF COMPOSITE STRUCTURAL CONCEPTS.

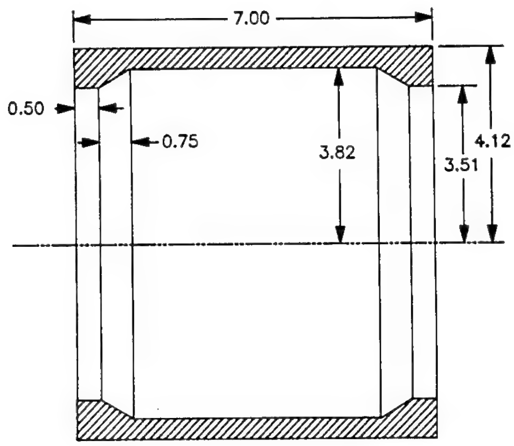
TECHNICAL APPROACH

- REVIEW LITERATURE/ON-GOING INVESTIGATIONS
- DEVELOP DYNAMIC MATERIAL TEST TECHNIQUES
- FABRICATE & DYNAMICALLY TEST MATERIAL SPECIMENS
- EVALUATE SPECIMEN FAILURES USING SEM
- ANALYZE MATERIAL SPECIMEN TESTS USING FEM
- ANALYZE GFRP CYLINDER USING FEM
- DYNAMICALLY TEST GFRP CYLINDER
- PROCURE STIFFENED/UNSTIFFENED MODELS FOR DYNAMIC TESTING

GFRP MODEL

- MATERIAL IS GRAPHITE/EPOXY (CELION 6K/E707)
- MANUFACTURED BY FILAMENT WINDING CIRCUMFERENTIALLY AND HAND LAYING UP LONGITUDINALLY
- $(90_2/0)_{32}$ LAY-UP WAS USED
- MODEL IS UNSTIFFENED AND WAS ORIGINALLY 0.61" THICK
- APES AND ADINA STATIC ANALYSES WERE USED TO OPTIMIZE STRUCTURAL MODIFICATIONS TO MODEL

GFRP-1 STRUCTURAL CONFIGURATION



DYNAMIC TESTING OF GFRP MODEL

- MODEL TESTED AT 4 INCREASING PRESSURE LOADINGS
- SHALLOW DEPTH (SHOCK WAVE ONLY)
- NDE CONDUCTED AFTER EACH TEST
- INSPECTION USING SEM AFTER FINAL TEST
- MODEL INSTRUMENTED WITH 5 BIAXIAL STRAIN GAGES

TEST GEOMETRIES

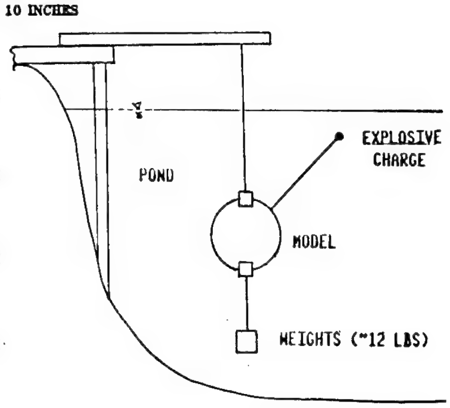
CHARGE WEIGHT = 4 GRAMS PENTOLITE
DEPTH OF CHARGE = 4 - 5 INCHES
MAXIMUM BUBBLE RADIUS = 10 INCHES

SHOT 1
STANDOFF = 2.4 FEET
PEAK PRESSURE = 1330 PSI

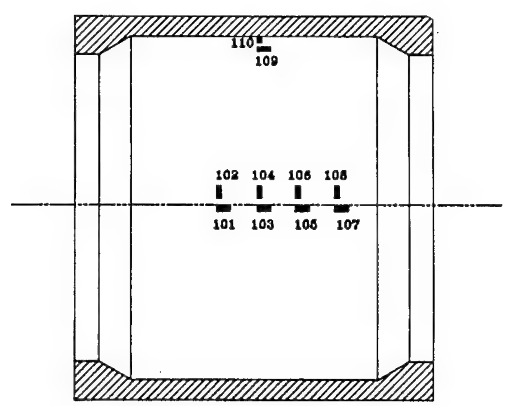
SHOT 2
STANDOFF = 1.2 FEET
PEAK PRESSURE = 3035 PSI

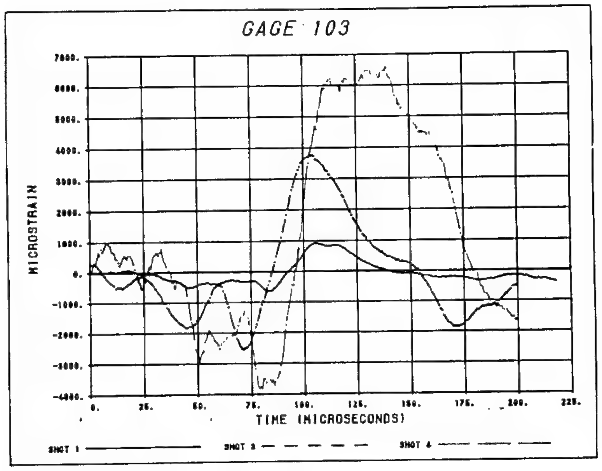
SHOT 3
STANDOFF = 0.67 FEET
PEAK PRESSURE = 6110 PSI

SHOT 4
STANDOFF = 0.51 FEET
PEAK PRESSURE = 8340 PSI



GFRP-1 INSTRUMENTATION LOCATION



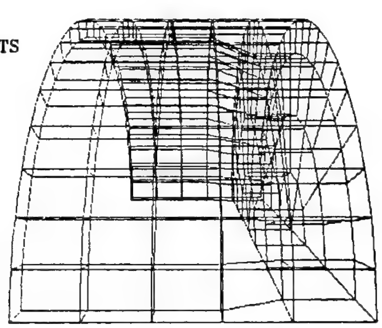


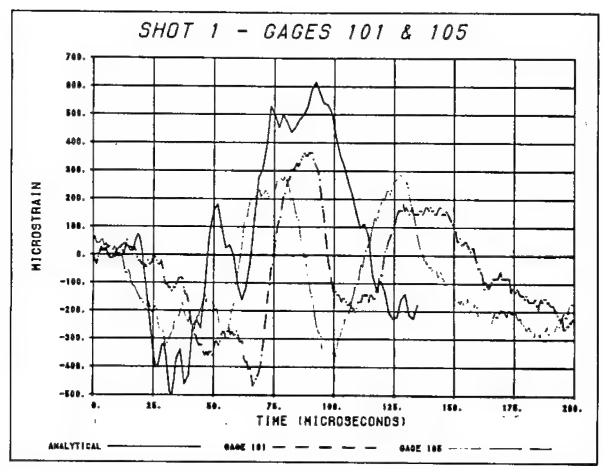
ADINA-S ANALYSIS

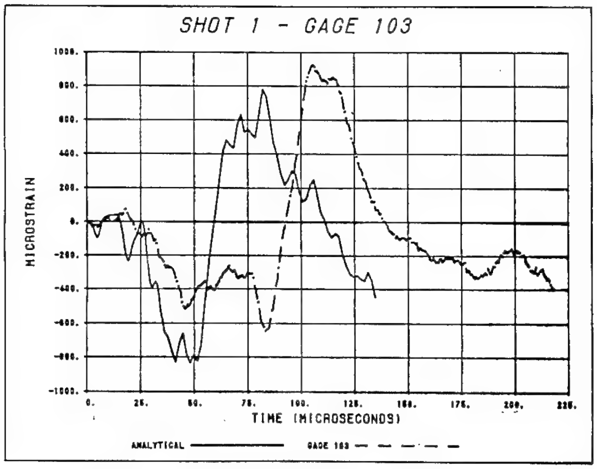
- SMEARED ORTHOTROPIC MATERIAL PROPERTIES USED INITIALLY (STATIC)
- UPDATED DYNAMIC MATERIAL PROPERTIES MAY BE INPUT LATER
- SPHERICAL INCIDENT WAVE (EXPONENTIAL TIME AND 1/STANDOFF DECAY) CALCULATED BY ADINA-S FOR GIVEN TEST GEOMETRY
- INTERACTION PRESSURE CALCULATED AT EACH TIME STEP USING DAA AND ADDED TO INCIDENT PRESSURE TO GIVE PRESSURE LOADING

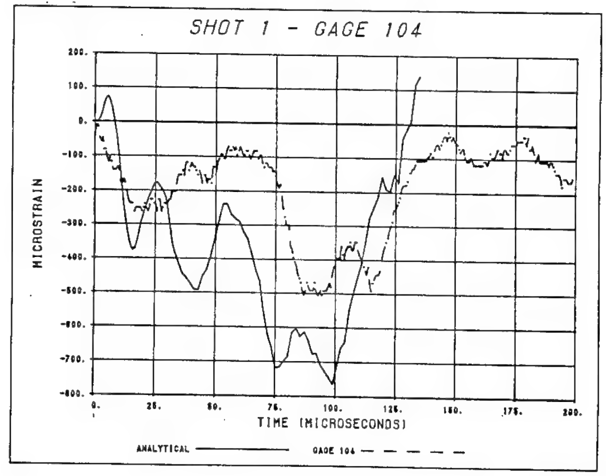
REFINED FEM MODEL

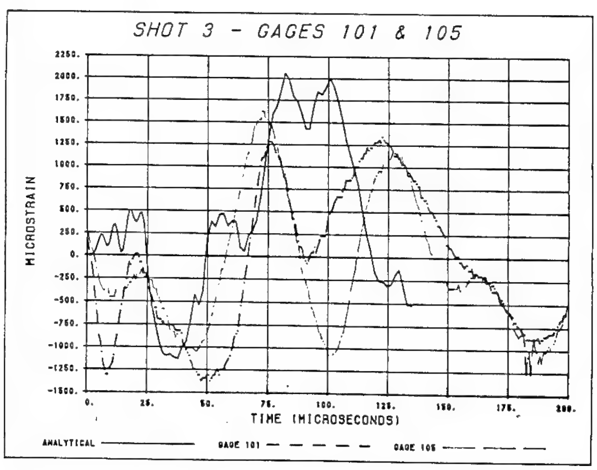
1130 NODES
141 3-D ELEMENTS
1942 D.O.F.

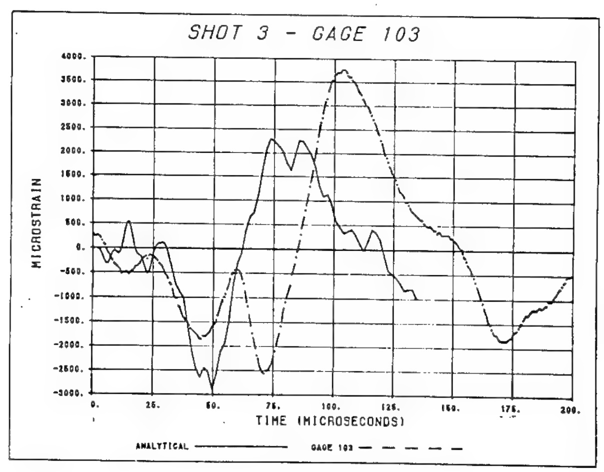


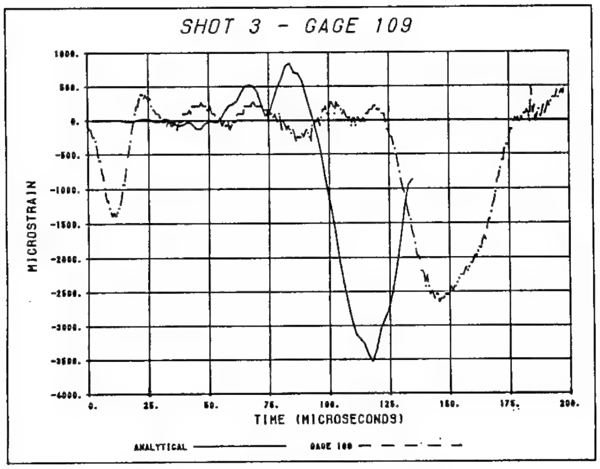


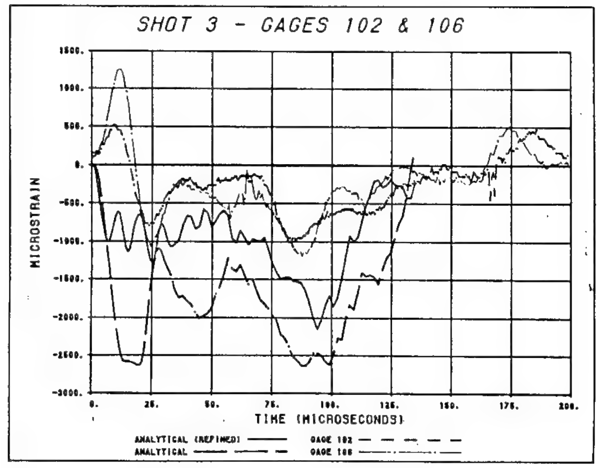

SHOT 1 - GAGE 103
MICROSTRAIN
TIME (MICROSECONDS)
ANALYTICAL
GAGE 103

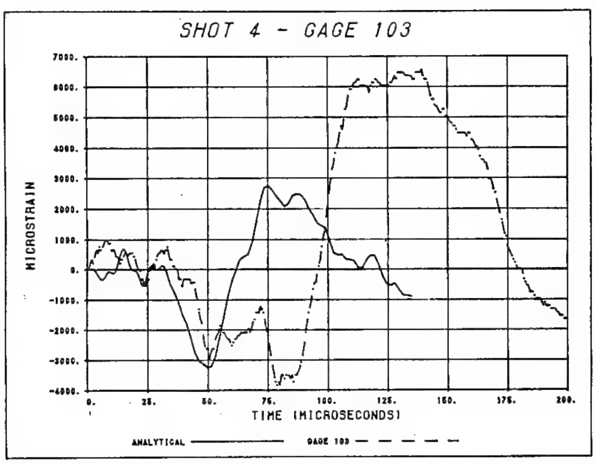

SHOT 1 - GAGE 104
MICROSTRAIN
TIME (MICROSECONDS)
ANALYTICAL
GAGE 104

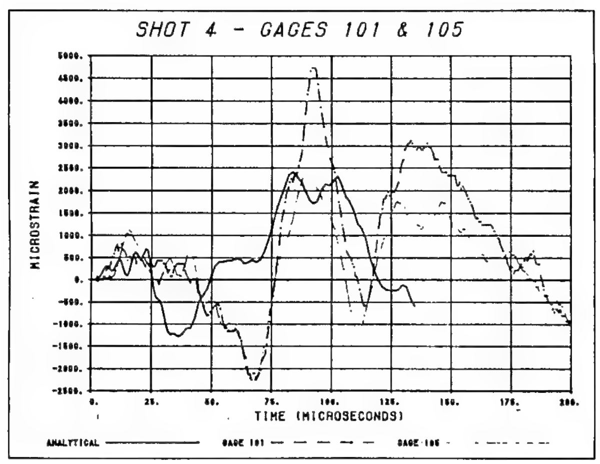

SHOT 3 - GAGES 101 & 105
MICROSTRAIN
TIME (MICROSECONDS)
ANALYTICAL
GAGE 101
GAGE 105


SHOT 3 - GAGE 103
MICROSTRAIN
TIME (MICROSECONDS)
ANALYTICAL
GAGE 103


SHOT 3 - GAGE 109
MICROSTRAIN
TIME (MICROSECONDS)
ANALYTICAL
GAGE 109


SHOT 3 - GAGES 102 & 106
MICROSTRAIN
TIME (MICROSECONDS)
ANALYTICAL (REFINED)
ANALYTICAL
GAGE 102
GAGE 106


SHOT 4 - GAGE 103
MICROSTRAIN
TIME (MICROSECONDS)
ANALYTICAL
GAGE 103


SHOT 4 - GAGES 101 & 105
MICROSTRAIN
TIME (MICROSECONDS)
ANALYTICAL
GAGE 101
GAGE 105

STATUS

- FOUR DYNAMIC TESTS PERFORMED ON GFRP MODEL
- FINITE ELEMENT ANALYSES CONDUCTED FOR EACH TEST GEOMETRY AND COMPARED TO EXPERIMENTAL DATA
- GOOD CORRELATION OBSERVED BETWEEN ANALYTICAL AND EXPERIMENTAL STRAIN DATA
- ULTRASONIC INSPECTION OF MODEL INDICATES SMALL AREA OF DAMAGE (DELAMINATION) INCURRED DURING FOURTH TEST
- MORE DETAILED ANALYSIS OF THROUGH THICKNESS RESPONSE AND DAMAGED AREA IS ONGOING

NEAR TERM PLANS

- THOROUGH NDE OF TESTED GFRP CYLINDER
- DESTRUCTIVELY EXAMINE GFRP CYLINDER OR POSSIBLE RETEST
- REFINE FEM ANALYSIS POSSIBLY INCLUDING EFFECTS OF DYNAMIC MATERIAL PROPERTIES
- PROCURE & TEST 8"-DIAMETER GRP CYLINDERS FOR COMPARISON WITH GRAPHITE/EPOXY RESULTS
- PROCURE LARGER DIAMETER GRAPHITE & GLASS MODELS TO INITIATE INVESTIGATION OF SCALING EFFECTS
- INITIATE INVESTIGATION OF THERMOPLASTIC RESIN SYSTEMS

DEFORMATION ANALYSIS OF COMPOSITE MATERIALS USING MOIRE INTERFEROMETRY

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Moire interferometry is a relatively new technique, but it has already been applied to numerous problems in experimental mechanics. While it has broad applicability, most of our work has been devoted to detailed studies of composite materials. This is a survey paper. The basic principles of moire interferometry are outlined and a number of recent applications to the study of composite material behavior are summarized. These applications include assessment of a new specimen configuration for in-plane shear property measurements, measurement of interlaminar properties of thick composites, determinations of thermal residual strains and mechanical residual strains, and detailed studies of edge-effect phenomena. Figures 2-9 address eight different studies.

Moire interferometry is an optical method. It provides whole-field contour maps of in-plane displacements with subwavelength sensitivity. The abundance of displacement data permits reliable determinations of normal strains and shear strains. The basic concepts are illustrated in Fig. 1. A crossed-line diffraction grating is replicated on the surface of the specimen. Coherent beams B_1 and B_2 interfere in their zone of intersection; parallel bands of constructive and destructive interference are formed and they act as a high frequency reference grating (a "virtual" reference grating). The specimen grating and reference grating interact to produce the moire fringe pattern seen in the camera. Analogous beams B_3 and B_4 in the vertical plane (not shown) create another reference grating which interrogates the second set of lines in the crossed-line specimen grating. By Eq. 2 (Fig. 1), patterns of N_x and N_y fringe orders are equivalent to contour maps of in-plane displacements U and V.

Normal strains and shear strains are determined from fringe gradients by Eqs. 3-5. Accidental rigid-body rotations are a well-known obstacle to practical utilization of moire techniques. However, this obstacle is circumvented by use of the 4-beam optical system, so the determination of shear strains is reliable.

Figure 2a introduces a new specimen configuration for measurement of in-plane shear properties. The displacement fields of (b) and (c) reveal a relatively large zone of uniform, pure shear. Normal strains ε_x and ε_y are practically zero. The specimen was a cross-ply laminate cut from a thick-walled cylinder. The shear-strain distribution between notches is plotted in (d), which shows nearly uniform shear strain in the test section. For this laminate, the result is slightly superior to the Iosipescu test. The specimen exhibited nonlinear shear behavior right from the start, as shown in (e). While laminate behavior might be calculated by laminate theory from the properties of unidirectional specimens, there is a persistent demand for direct measurement of laminate properties.

An investigation of interlaminar shear is illustrated in Fig. 3. The specimen was cut from an 8-inch diameter thick-walled cylinder specified in (a). The two rail-shear configurations illustrated in (b) were used, each with the composite bonded to steel rails. The V displacement field is represented by the fringe pattern in (c). A special carrier-fringe technique was employed here to transform the pattern to one in which the local displacement gradients are determined from the local fringe slopes. Detailed micromechanical behavior (on the ply level) can be extracted by careful examination. The graph in (d) shows the interlaminar shear strain distribution along the length of individual 0° and 90° plies. Graph (e) depicts the shear strain distribution across the width of the specimen (at mid-length); the insert shows the locations of 0° and 90° plies in the specimen. The 90° plies exhibit greater shear compliance than the 0° plies, but resin-rich zones between plies are most compliant. These phenomenological results are dramatic evidence of the statistical variation of performance, where nominally equivalent plies exhibit different deformations. Average values of interlaminar shear moduli, proposed as suitable for design calculations for this material, are given in (f).

Figure 4 illustrates an investigation of the same material under interlaminar compression. The test configuration is depicted in (a). The U, V and W displacement fields are shown in (b) for one face. The U field, shown for a higher load level in (c), reveals a strong edge effect -- a local interlaminar shear. Its distribution is plotted in (d); again, the ply-by-ply variation is evident. The corresponding interlaminar shear stresses were calculated, using properties listed in Fig. 3(f), and plotted in (e). The W field shows the ridges and furrows of the deformed specimen surface. The V field gave perplexing results and this issue is revisited in Fig. 6.

As illustrated in Fig. 5, the residual thermal strains induced by cooling specimen (a) from its cure temperature to room temperature were investigated. The deformations strongly resembled those of

interlaminar compression. Again an edge effect -- or surface skin effect -- was evident on faces A and B. Interlaminar shear strains reached very high levels near free surfaces, as seen in (d).

Figure 6 illustrates an on-going study of the bimaterial interface problem. Deformations induced by a 240°F change of temperature of specimen (A) were revealed by the fringe patterns of (b), (c) and (d). The normal stress σ_y across line A-A was determined from these patterns and known material properties. The results in (e) show severe gradients and high stresses near the interface. An interfacial region less then 0.005 inches thick is represented by the broad line, where the shape of the stress curve remains undetermined. Companion 3-D finite-element analyses indicate that the curve hooks sharply in the brass and extends toward the stress in the steel. Such bimaterial interface stresses must abound near free edges of composite laminates, too.

Residual strains in a thick composite cylinder of $[90_2/0]_{27}$ ply sequence are revealed in Fig. 7. The technique, illustrated in (b) was to relieve the residual stresses in the vicinity of the specimen grating by slicing and undercutting. Graph (d) shows very high radial strains (ε_x) that tend to cause delaminations. It also shows two regions of high shear strains, and these correspond to locations of intermediate curing cycles in the fabrication process.

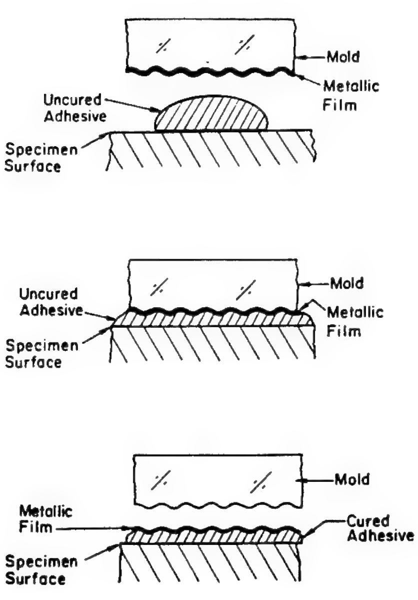
Figure 8(a) defines two metal-matrix specimens. The published analysis of Spec. I was followed by a study of Spec. II, which had a 45° fiber direction in its outer ply. Anomalous effects revealed in (b) include movement of groups of fibers relative to neighboring groups. Graph (c) shows intra-fiber shear strains on a micromechanical level and also on a continuum (average) basis. The shear strains are concentrated in the matrix material. Disproportionately large shears occur at random locations. Strong interlaminar shears must accompany these fiber movements.

Figure 9 illustrates a unique micromechanics study currently in progress. A miniature moire interferometer was developed to measure deformations inside a hole in a laminated compression panel.

Attributes of moire interferometry implicit in the above applications are outlined in Fig. 10. The method is amenable to a wide range of mechanics problems -- macro and micromechanics. Various material systems can be investigated. It is applicable for phenomenological studies, measurement of material properties, assessment of test methods, engineering analyses and verification of theory. Future extensions contemplate a ten-fold increase of sensitivity for micromechanics studies on a still finer scale.

Grateful acknowledgement for support of this work is extended to the Office of Naval Research (Grant N00014-86-K0255), to the National Center for Composite Materials Research, Urbana, Illinois (subcontract under ONR Grant N00014-86-K0799) and NASA Langley Research Center (Grant NAG-1-343).

Fig. 1 Moiré interferometry



(a) Replicate a crossed-line high-reflectance grating on the specimen. Grating frequency is 1200 lines/mm.

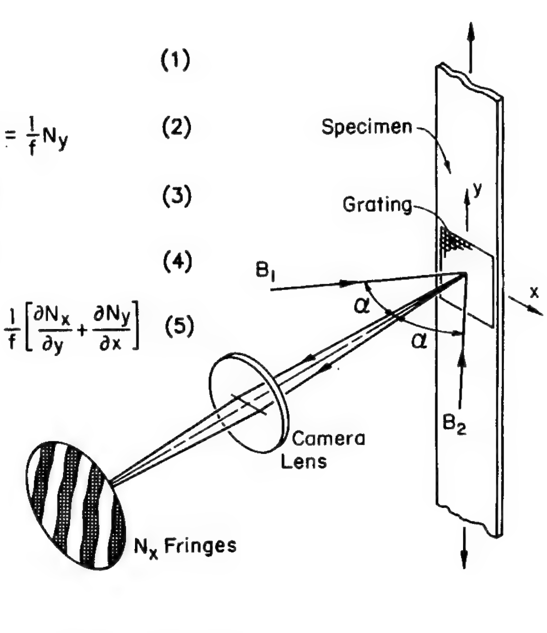
$$f = \frac{2}{\lambda}\sin\alpha \quad (1)$$

$$U = \frac{1}{f}N_x \qquad V = \frac{1}{f}N_y \quad (2)$$

$$\epsilon_x = \frac{\partial U}{\partial x} = \frac{1}{f}\frac{\partial N_x}{\partial x} \quad (3)$$

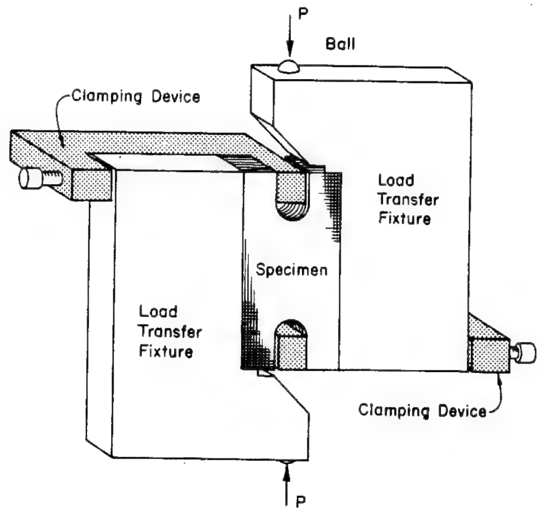
$$\epsilon_y = \frac{\partial V}{\partial y} = \frac{1}{f}\frac{\partial N_y}{\partial y} \quad (4)$$

$$\gamma_{xy} = \frac{\partial U}{\partial y} + \frac{\partial V}{\partial x} = \frac{1}{f}\left[\frac{\partial N_x}{\partial y} + \frac{\partial N_y}{\partial x}\right] \quad (5)$$

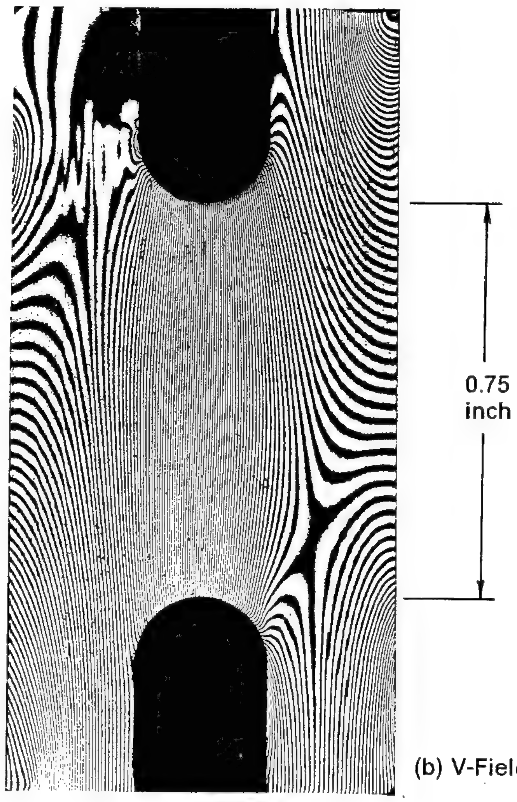


(b) Schematic of optical system for moiré interferometry. Beams B_3 and B_4 in vertical plane (not shown) form the N_y pattern. f = 2400 lines/mm = 60,960 lines/in.

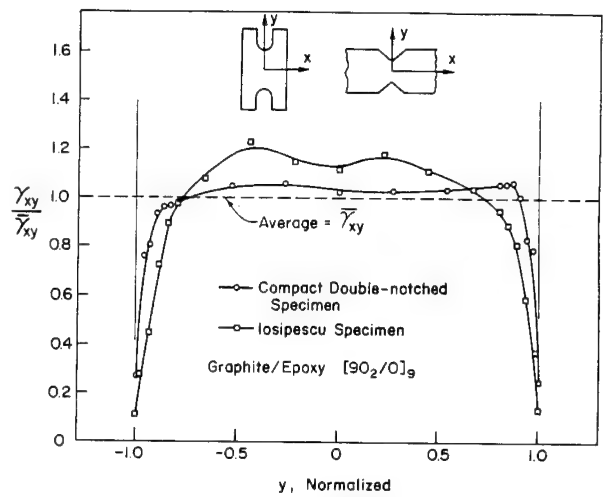
Key words: Fig. 2
Compact double-notched shear specimen, In-plane shear, Test Methods, Pure shear, Thick Graphite/Epoxy $[90_2/0]_{27}$



(a) Specimen and Fixture

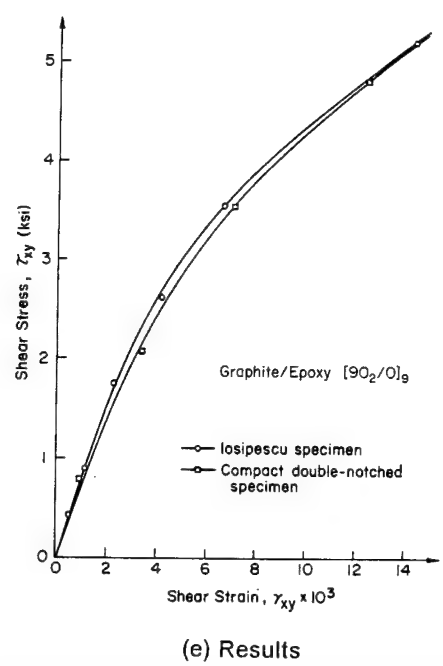


(b) V-Field



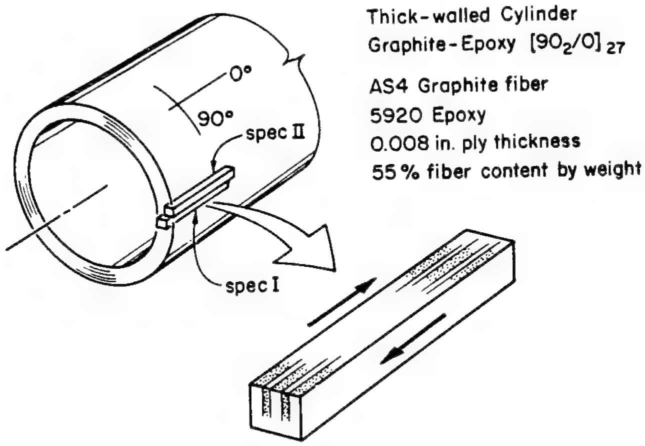
(d) Strain Distribution

(c) U-Field

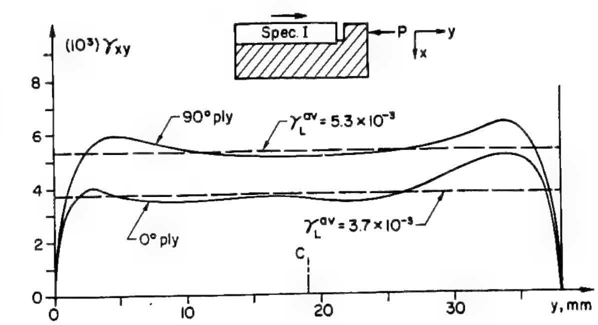


(e) Results

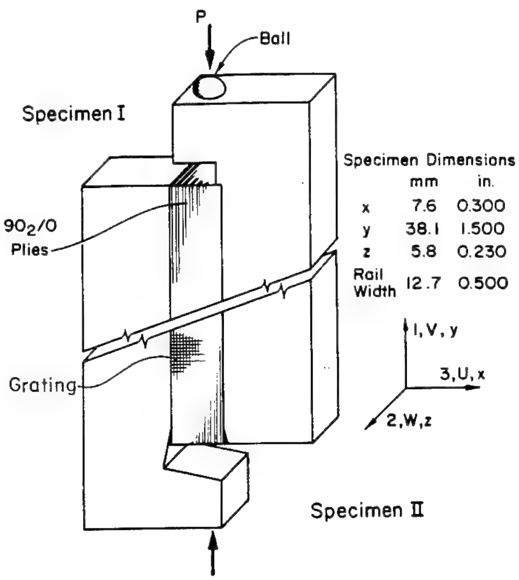
Key words: Fig. 3
Interlaminar shear, Thick Gr/Ep $[90_2/0]_{27}$ Phenomenological behavior, Representative shear moduli



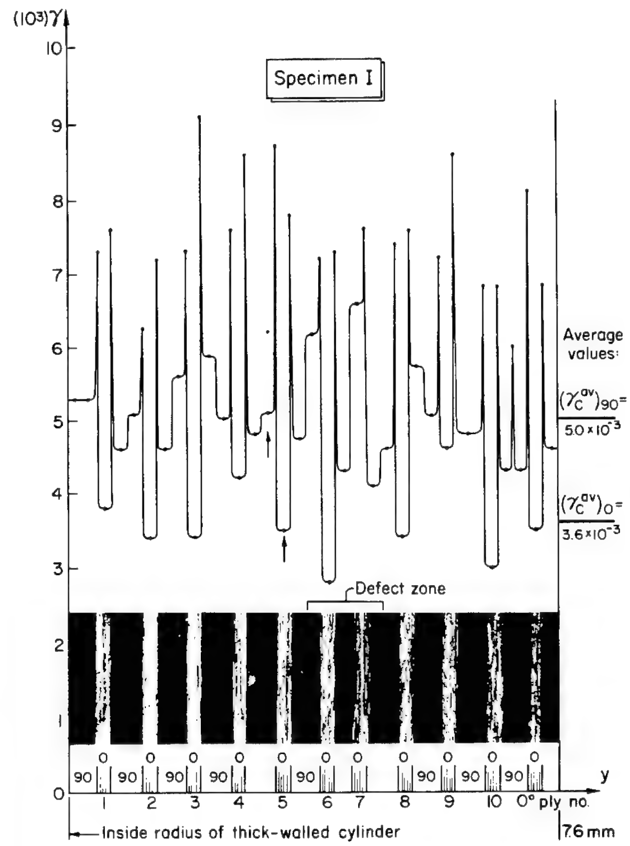
(a) Specimen



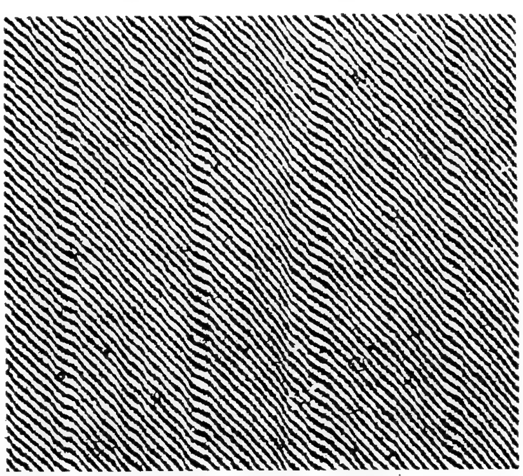
(d) γ_{xy} along individual plies



(b) Experimental method



(e) γ_{xy} along the width of the specimen



(c) V-field (central region)

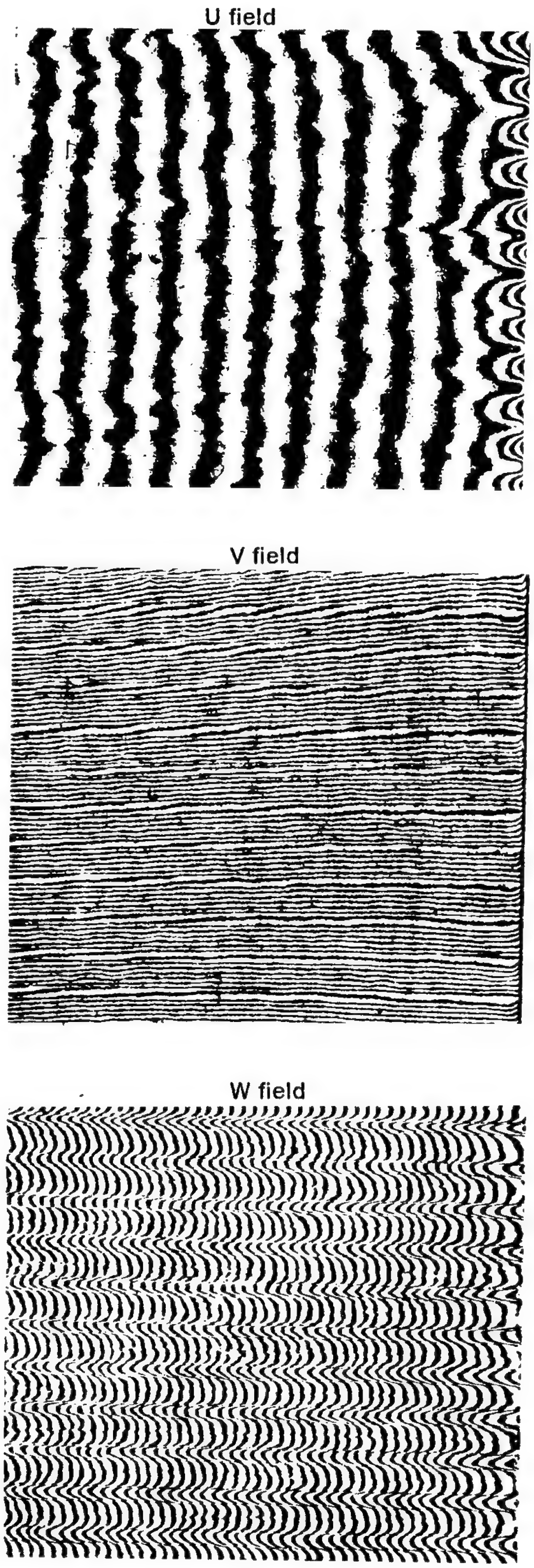
G_{13} = 3.2 GPa = 470,000 psi

G_{23} = 2.4 GPa = 350,000 psi

$G^{eff}_{[90_2/0]_n}$ = 2.6 GPa = 380,000 psi

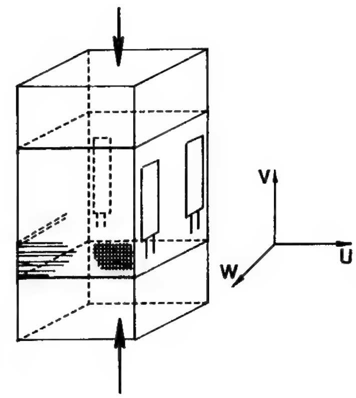
(f) Interlaminar shear moduli (average values in specimen)

Key words: Fig. 4
Interlaminar compression, Thick Gr/Ep $[90_2/0]_{27}$, Edge effects, Interlaminar shear

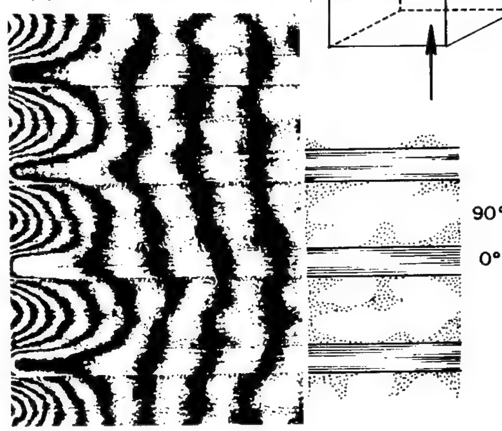


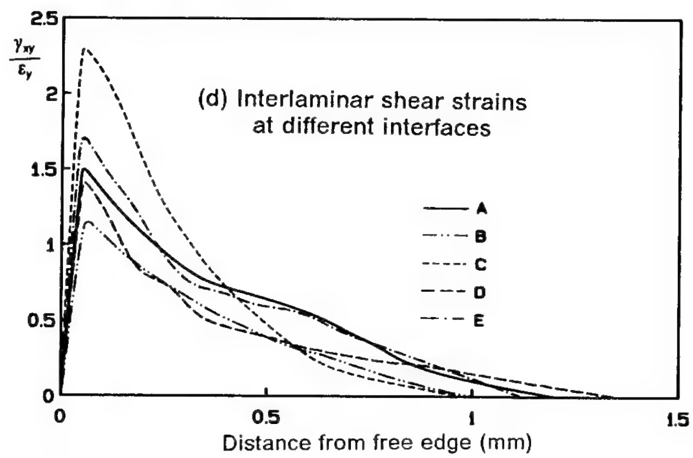
(b) Displacement fields U, V, W

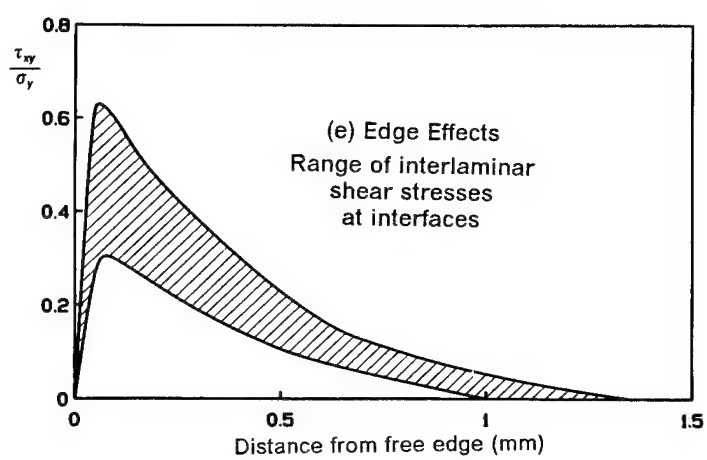
(a) Specimen with gratings and strain gages



(c) U, for higher load

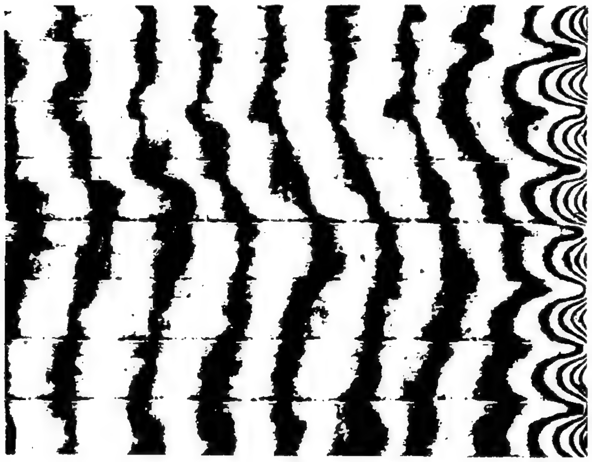




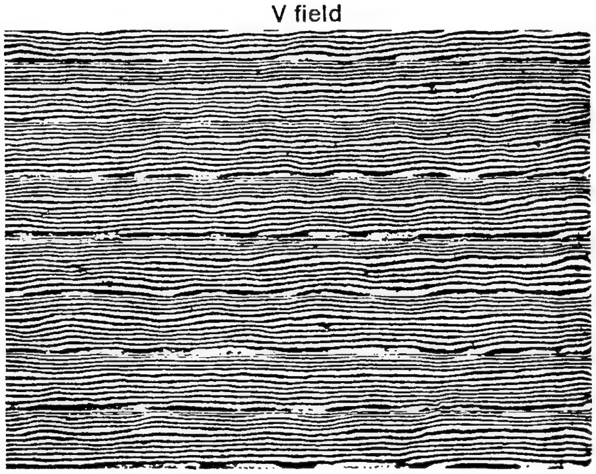


Key words: Fig. 5
Residual thermal strains, Thick Gr/Ep $[90_2/0]_{27}$, Edge effects, Test method

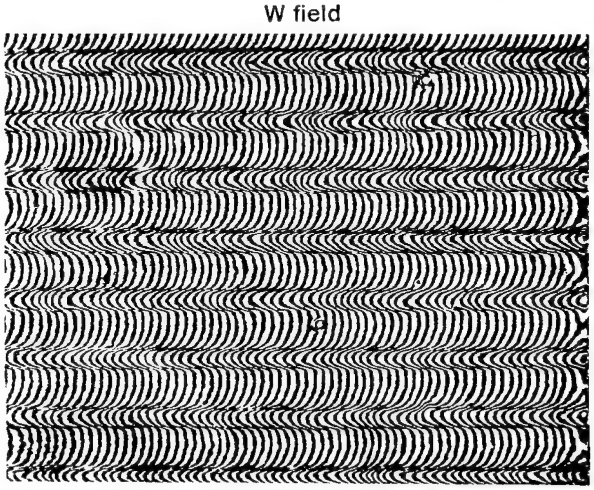
U field



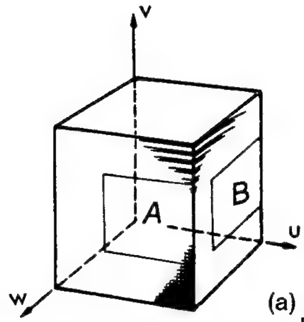
V field



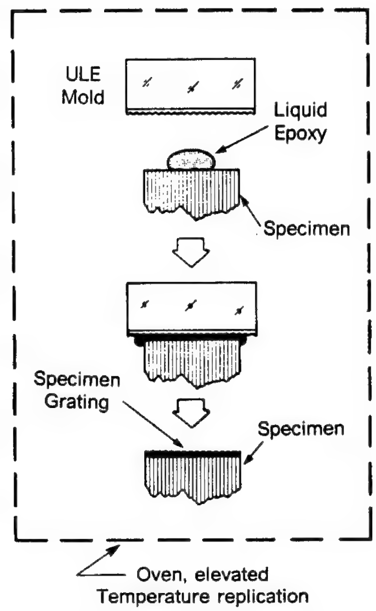
W field



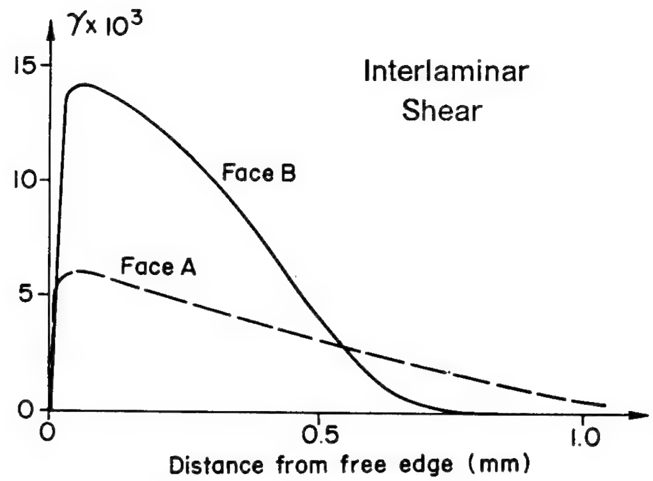
(c) Residual deformation at room temp.
Face A, $\triangle T = 180°F$



(a) Specimen
$[90_2/0]_{27}$,

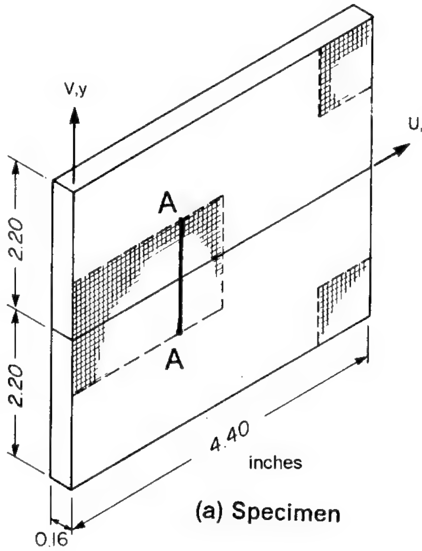


(b) Grating applied at cure temp., 250°F

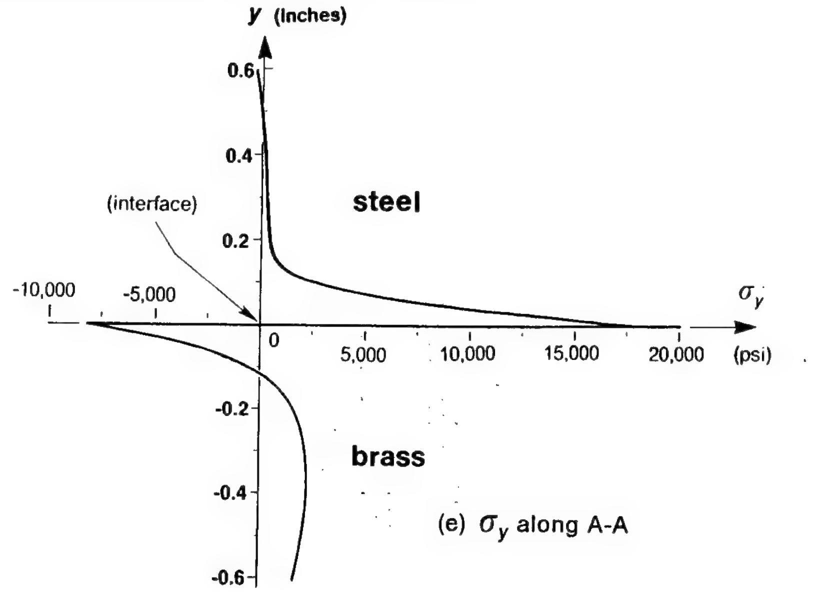


(d) Edge effects

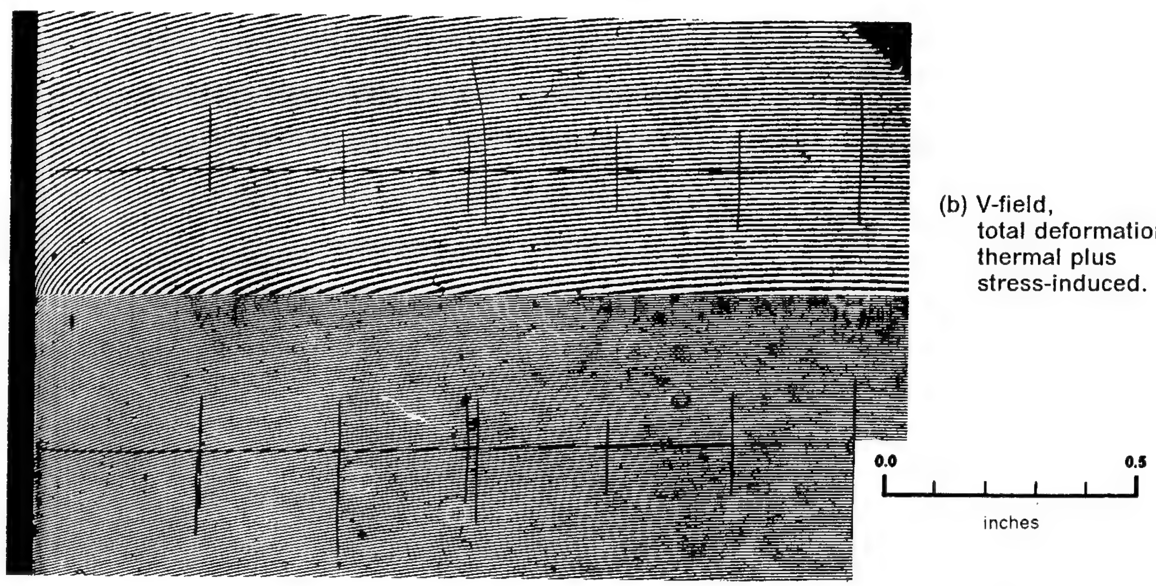
Key words: Fig. 6
Interface problem, Thermal stresses, Bimaterial joint, Line singularity, 3-D effects.



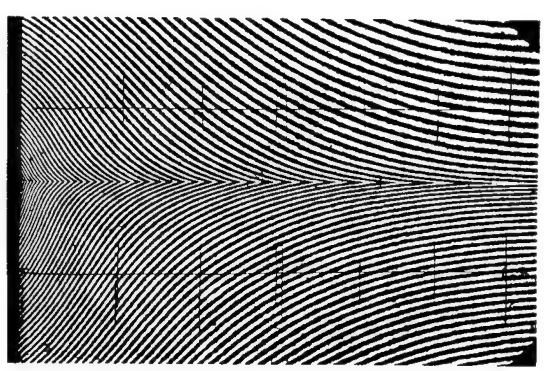
(a) Specimen

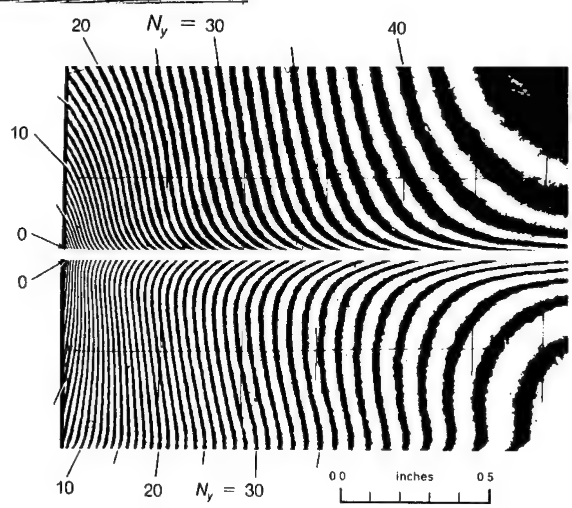


(e) σ_y along A-A



(b) V-field,
total deformation,
thermal plus
stress-induced.

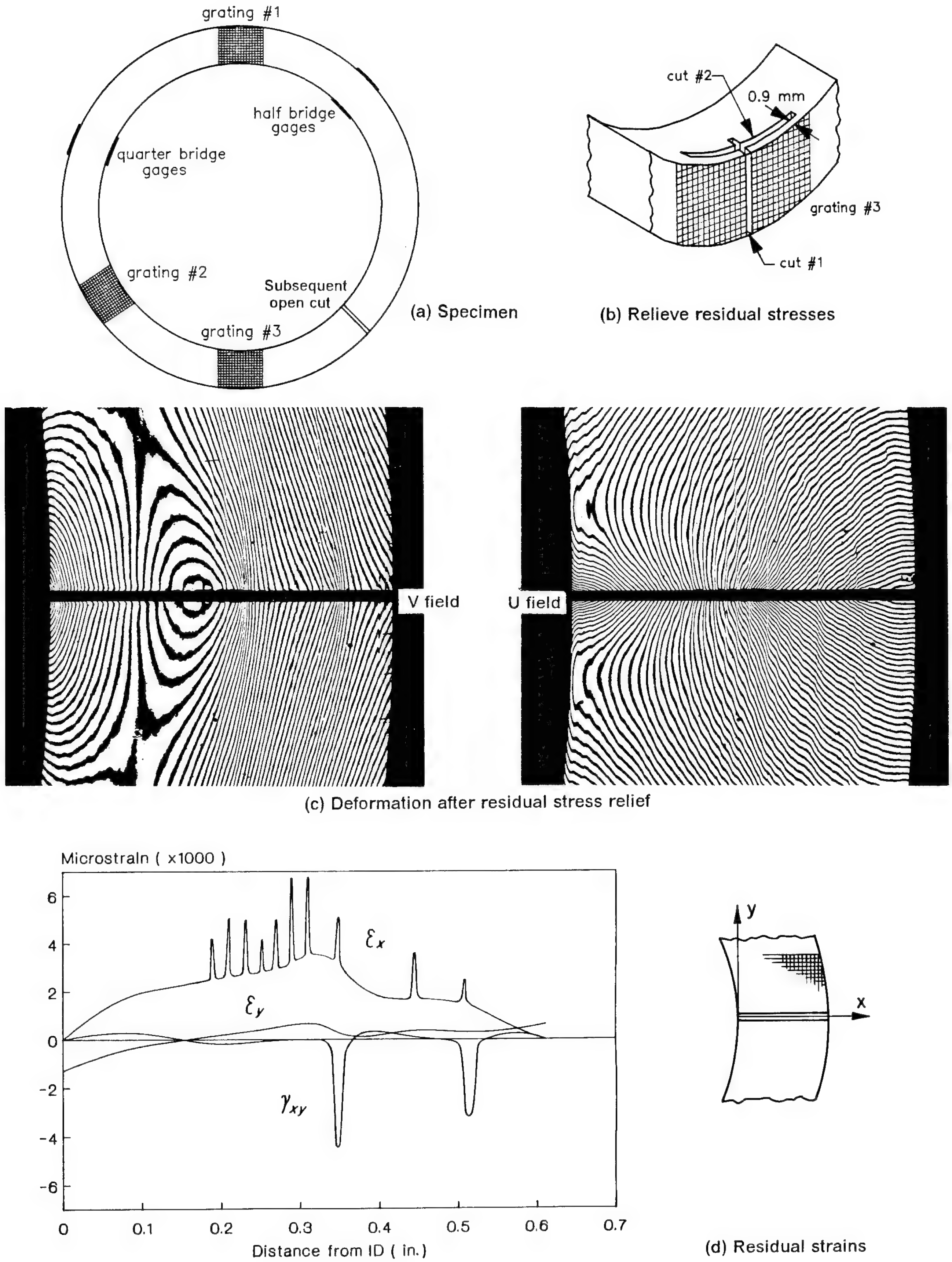




(c) V, with carrier

(d) V, Stress-induced part

Key words: Fig. 7
Residual strains, Thick composite, Gr/Ep $[90_2/0]_{27}$



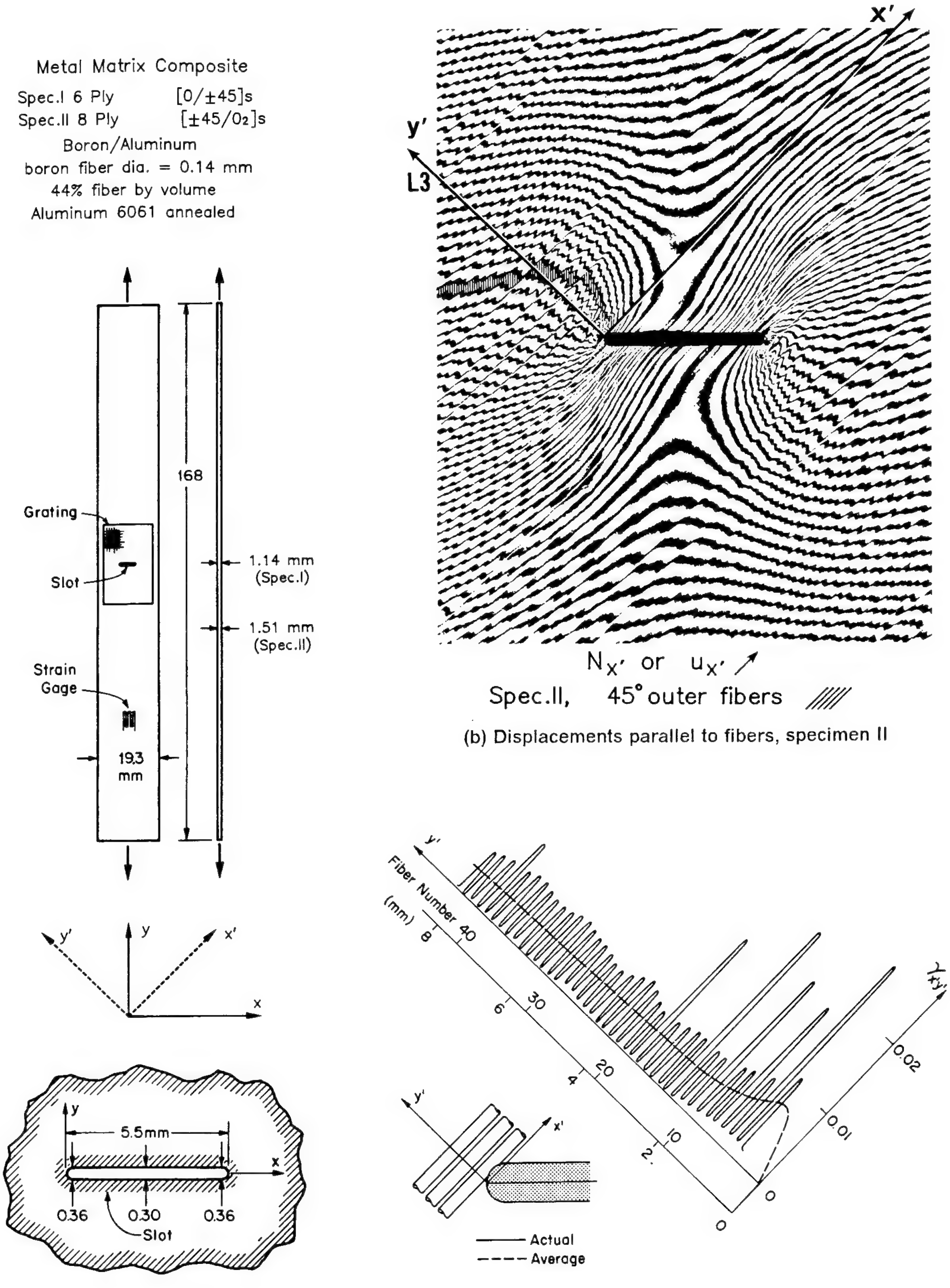
(a) Specimen

(b) Relieve residual stresses

(c) Deformation after residual stress relief

(d) Residual strains

Key words: Fig. 8
Metal-matrix laminate, Micromechanics, 45°ply, Intra-fiber shear, Interlaminar shear.

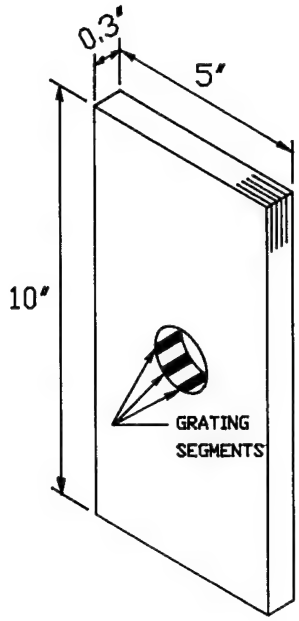


(a) Specimen and material

(b) Displacements parallel to fibers, specimen II

(c) Shear strain along line L3

Key words: Fig. 9
Edge effects in holes, Micromechanics, Thick laminate, Compression



STACKING SEQUENCE

$[0_2/90]_{5S}$

$[0/45/90/-45/0]_{3S}$

HOLE SIZE

1' DIAMETER CIRCULAR HOLE

1' x 1-1/2' ELLIPTICAL HOLE

MAXIMUM LOAD

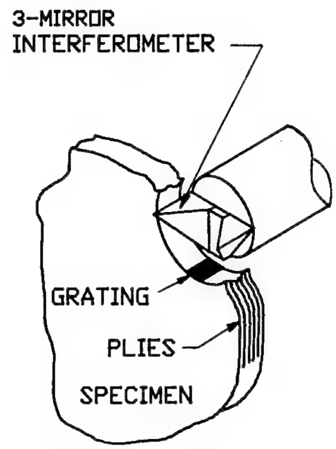
60,000 lbs. COMPRESSION

MATERIAL

GLASS/EPOXY

GRAPHITE/EPOXY

(a) Experimental parameters



(b) Optics in hole

Fig. 10

CONCLUSIONS

Optical Method
- In-plane displacements
- High signal-to-noise
- High spatial resolution
- High sensitivity

Applications
- Macromechanics
- Micromechanics
- Thermal stains
- Residual strains

Measurements
- Displacements (U,V)
- Normal strains (ε)
- Shear strains (γ)

APPENDIX A: PROGRAM LISTINGS

AIR FORCE OFFICE OF SCIENTIFIC RESEARCH

INHOUSE

NONE

GRANTS AND CONTRACTS

DAMAGE MODELS FOR CONTINUOUS FIBER COMPOSITES
AFOSR-84-0067
01 April 87 - 31 March 88

Principal Investigators: Dr David Allen
Dr Charles E Harris
Department of Aerospace Engineering
Texas A+M University
College Station, TX 77843
(409) 845-7541

Program Manager: Lt Colonel George K Haritos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-0463

Objective: To develop a damage model for predicting strength and stiffness of continuous fiber composite structure subjected to fatigue loading, and to verify this model with experimental results.

DIRECT OBSERVATION OF CRACKING AND THE DAMAGE MECHANICS OF CERAMICS AND CERAMIC COMPOSITES
AFOSR-87-0307
01 June 87 - 31 May 90

Principal Investigator: Dr Peter W. R. Beaumont
Engineering Department
Cambridge University
Trumpington Street, Cambridge CB2 1PZ
(01144) 223-332600

Program Manager: Lt Colonel George K Haritos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-0463

Objective: To directly observe and analyze microcracking and spalling in ceramic materials. To study inherent toughening methods, such as plasticity in oxide ceramics at high temperature, constrained plasticity, soft cobalt matrices, and localized transformation toughening.

STUDIES IN THE DELAMINATION FRACTURE BEHAVIOR OF COMPOSITE MATERIALS
AFOSR-84-0064
01 August 87 - 31 July 89

Principal Investigator: Dr Walter L Bradley
Department of Mechanical Engineering
Texas A+M University
College Station, TX 77843
(409) 845-1259

Program Manager: Lt Colonel George K Haritos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-0463

Objective: To illucidate the micromechanisms underlying delamination failures in polymer composites which are subjected to mixed-mode loading.

ELEVATED TEMPERATURE PERFORMANCE OF CERAMIC AND GLASS MATRIX COMPOSITES
AFOSR-87-0383
15 July 87 - 14 October 91

Principal Investigator: Professor Tsu-Wei Chou
University of Delaware
Center for Composite Materials
Newark, DE 19716
(302) 451-2904

Program Manager: Dr Liselotte J. Schioler
AFOSR/NE
Bolling AFB DC 20332-6448
(202) 767-4933

Objective: To provide a fundamental understanding of the high-temperature mechanical properties, environmental effects and failure mechanisms of glass and ceramic matrix composites through experimental characterization and theoretical modeling, and to establish high-temperature mechanical testing and characterization methods for glass and ceramic matrix composites.

HETEROGENEOUS CHARACTERIZATION OF COMPOSITE MATERIALS WITH PROGRESSIVE DAMAGE
AFOSR-88-0124
01 February 88 - 31 January 91

Principal Investigator: Dr Isaac M. Daniel
Department of Civil Engineering
Northwestern University
Evanston, IL 60201
(312) 491-5649

Program Manager: Lt Colonel George K Haritos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-0463

Objective: To develop constitutive and failure models for composite materials based on observed damage mechanisms and damage development. The study will include organic matrix composite materials such as graphite/epoxy, as well as high-temperature composites, such as ceramic matrix/ceramic fiber composites.

MICROCRACKING AND TOUGHNESS OF CERAMIC-FIBER/CERAMIC-MATRIX COMPOSITES UNDER HIGH TEMPERATURE
AFOSR-87-0288
01 August 87 - 30 July 89

Principal Investigators: Dr Feridun Delale
Dr Been-Ming Liaw
Department of Mechanical Engineering
The City College of
The City University of New York
New York, NY 10036
(212) 690-4252

Program Manager: Lt Colonel George K Haritos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-0463

Objective: To study the mechanisms of microcracking at the fiber/matrix level in a ceramic-fiber/ceramic-matrix composite material subjected to thermomechanical loading.

DYNAMICS AND AEROELASTICITY OF COMPOSITE STRUCTURES
F49620-86-C-0066
01 July 86 - 30 June 90

Principal Investigator: Dr John Dugundji
Department of Aeronautics + Astronautics
Massachusetts Institute of Technology
Cambridge, MA 02139
(617) 253-3758

Program Manager: Dr Anthony K Amos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To pursue combined experimental and theoretical investigations of aeroelastic tailoring effects on flutter and divergence of aircraft wings.

DEFORMATION AND DAMAGE MECHANISMS IN HIGH TEMPERATURE COMPOSITES WITH DUCTILE MATRICES
AFOSR-88-0150
01 March 88 - 28 February 91

Principal Investigator: Dr George J Dvorak
Department of Civil Engineering
Rensselaer Polytechnic Institute
Troy, NY 12181
(518) 266-6943

Program Manager: Lt Colonel George K Haritos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-0463

Objective: To develop a basis of understanding of the thermo-mechanical behavior of fibrous composites with ductile matrices and ductile or elasto-brittle fibers, and of the damage mechanisms activated by combined mechanical and thermal loading, both cyclic and monotonic.

FAILURE OF LAMINATED PLATES CONTAINING HOLES
AFOSR-87-0204
01 April 87 - 31 March 89

Principal Investigator: Dr E S Folias
Department of Civil Engineering
The University of Utah
Salt Lake City, UT 84112
(801) 581-6931

Program Manager: Lt Colonel George K Haritos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-0463

Objective: To analytically determine the three-dimensional stress field in a laminated plate containing a cylindrical hole through its entire thickness and loaded uniformly in the in-plane direction and to establish failure criteria.

PREDICTION AND CONTROL OF PROCESSING-INDUCED RESIDUAL STRESSES IN COMPOSITES
AFOSR-87-0242
01 June 87 - 31 May 89

Principal Investigator: Dr H Thomas Hahn
Department of Engineering Science and Mechanics
The Pennsylvania State University
University Park, PA 16802
(814) 863-0997

Program Manager: Lt Colonel George K Haritos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-0463

Objective: To identify the mechanisms underlying the introduction of residual stresses during processing of polymer matrix composites, and to develop a prediction methodology as well as a procedure for controlling these stresses through optimization of the process cycle.

MODELING OF THE IMPACT RESPONSE OF FIBRE-REINFORCED COMPOSITES
AFOSR-87-0129
15 November 86 - 14 November 89

Principal Investigators: Dr John Harding
Dr C Ruiz
Department of Engineering Science
University of Oxford
Oxford, OX1 3PJ England

Program Manager: Dr Anthony K Amos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To characterize the mechanical behavior and failure mechanisms of carbon/epoxy, Kevlar/epoxy, and hybrid composites under tensile impact loading using specially designed split Hopkinson bar equipment.

CRAZING IN POLYMERIC AND COMPOSITE SYSTEMS
AFOSR-87-0143
01 April 87 - 31 March 90

Principal Investigator: Dr C C Hsiao
Deparment of Aerospace Engineering and Mechanics
University of Minnesota
Minneapolis, MN 55455
(612) 625-7363

Program Manager: Lt Colonel George K Haritos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-0463

Objective: To develop time-dependent theories for the crazing behavior of polymeric and structural composite systems by understanding the microstructural behavior of the materials during crazing.

STRUCTURE AND PROPERTIES OF HIGH SYMMETRY CERAMIC MATRIX COMPOSITES
AFOSR-88-0075
01 December 1987 - 30 November 1990

Principal Investigator: Professor Frank Ko
Dept of Materials Engineering
Drexel University
Philadelphia, PA 19104
(215) 895-1640

Program Manager: Dr Liselotte J Schioler
AFOSR/NE
Bolling AFB DC 20332-6448
(202) 767-4933

Objective: To determine the merit of a high symmetry hybrid architecture consisting of ceramic spheres, kevlar fibers and brittle matrix for creating tough solids by modelling, processing, optimization and mechanical properties evaluation.

THE MECHANICS OF PROGRESSIVE CRACKING IN CERAMIC MATRIX COMPOSITES AND LAMINATES
AFOSR-88-0104
01 February 88 - 31 January 91

Principal Investigator: Dr Norman Laws
Dept of Mechanical Engineering
University of Pittsburgh
Pittsburgh, PA 15261
(412) 624-9793

Program Manager: Lt Colonel George K Haritos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-0463

Objective: To study damage processes in continuous fiber-reinforced ceramic matrix composites (CMC), and, in particular, the degradation (or improvement) of thermo-mechanical properties when the composites have been damaged by matrix cracking, fiber debonding, and ultimately fiber pull-out.

OPTIMUM AEROELASTIC CHARACTERISTICS FOR COMPOSITE SUPER-MANEUVERABLE AIRCRAFT
F49620-87-C-0046
01 June 87 - 31 May 88

Principal Investigator: Dr Gabriel Oyibo
Department of Mechanical + Aerospace Engineering
Polytechnic University
Farmingdale, NY 11735
(516) 454-5120

Program Manager: Dr Anthony K Amos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To identify, characterize, and model the effects of constrained warping on the dynamics and aeroelastic stability of aircraft composite wings.

MICROMECHANICAL ANALYSIS OF CERAMIC MATRIX COMPOSITES
F49620-88-C-0069
01 April 88 - 31 March 90

Principal Investigator: Dr B Walter Rosen
Materials Sciences Corporation
Gwynedd Plaza II
Bethlehem Pike
Spring House, PA
(215) 542-8400

Program Manager: Lt Colonel George K Haritos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-0463

Objective: To develop a material model of a unidirectional composite which accounts for residual stresses, matrix porosity, interphases, cracks perpendicular to fibers, cracks parallel to fibers, interface debonding, fiber fracture, and in general, the accumulation and growth of various types of damage.

STUDIES ON DEFORMATION AND FRACTURE OF VISCOELASTIC COMPOSITE MATERIALS
AFOSR-87-0257
01 July 87 - 30 June 89

Principal Investigator: Dr Richard A Schapery
Department of Civil Engineering
Texas A+M University
College Station, TX 77843
(409) 845-2449

Program Manager: Lt Colonel George K Haritos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-0463

Objective: To develop and verify mathematical models of deformation and delamination of viscoelastic composites with distributed micro-damages.

CONTROL AUGMENTED STRUCTURAL OPTIMIZATION OF AEROELASTICALLY TAILORED FIBER COMPOSITE WINGS
F49620-87-K-0003
01 November 86 - 31 October 89

Principal Investigators: Dr Lucien A Schmit
Dr Peretz Friedmann
Dept of Mechanical, Aerospace and Nuclear Engineering
University of California, Los Angeles
Los Angeles, CA 90024
(213) 825-7697

Program Manager: Dr Anthony K Amos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To develop a control-augmented optimization capability for the efficient aeroelastic tailoring of composite wings and lifting surfaces. The analytical methods to be developed should permit extension of formal optimization procedures in design applications beyond current capabilities.

PREPARATION AND CHARACTERIZATION OF CARBON AND Si(C) FILAMENTS
F49620-86-C-0083
15 November 87 - 14 October 89

Principal Investigator: Professor Ian L Spain
Department of Physics
Colorado State University
Fort Collins, CO 80523
(303) 491-6076

Program Manager: Dr Liselotte J Schioler
AFOSR/NE
Bolling AFB DC 20332-6448
(202) 767-4933

Objective: To establish the principles governing the growth of carbon and silicon carbide filaments and to study the interrelationships of crystal structure, microstructure and electrical and mechanical properties of these microscale materials.

A STUDY OF THE CRITICAL FACTORS CONTROLLING THE SYNTHESIS OF CERAMIC MATRIX COMPOSITES FROM PRECERAMIC POLYMERS
F49620-87-C-0093
15 September 87 - 14 August 90

Principal Investigator: Professor James R Strife
United Technologies Research Center
East Hartford, CT 06108
(203) 727-7270

Program Manager: Dr Liselotte J Schioler
AFOSR/NE
Bolling AFB DC 20332-6448
(202) 767-4933

Objective: To investigate the critical factors which determine the mechanical properties of composites synthesized from a preceramic polymer matrix and carbon or ceramic fibers.

A COMPREHENSIVE STUDY ON MICROSTRUCTURE-MECHANICS RELATIONSHIPS OF CERAMIC MATRIX COMPOSITES
AFOSR-88-0113
01 April 88 - 31 March 89

Principal Investigator: Dr Albert S D Wang
Department of Mechanical Engineering and Mechanics
Drexel University
Philadelphia, PA 19104
(215) 895-2297

Program Manager: Lt Colonel George K Haritos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-0463

Objective: To establish both a fabrication and a material characterization capability for a class of high-temperature ceramic matrix composites as an integrated effort.

INVESTIGATIONS OF THERMALLY-INDUCED DAMAGE IN COMPOSITES
AFOSR-87-0128
01 March 87 - 29 February 88

Principal Investigator: Dr Y Weitsman
Department of Civil Engineering
Texas A+M University
College Station, TX 77843
(409) 845-7512

Program Manager: Lt Colonel George K Haritos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-0463

Objective: To develop a constitutive model, generic to the response of composite materials under load, temperature, and moisture. Special emphasis is placed on the evolution of damage within the material under the influence of interacting drivers such as the diffusion of moisture and temperature.

U. S. ARMY
ARMY RESEARCH OFFICE

CONTRACTS

TITLE: Basic Research into Static and Dynamic Properties of Composite Blades with Structural Couplings
RESPONSIBLE INDIVIDUAL G. L. Anderson
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641
PRINCIPAL INVESTIGATOR: John Dugundjii
Dept. of Aeronautics and Astronautics
Massachusetts Institute of Technology
Cambridge MA 02139

OBJECTIVE: Study analytically and experimentally the static and dynamic behavior of helicopter rotor blades made of composite materials.

TITLE: Advanced Mechanical Design of High Performance Articulating Robotic Systems
RESPONSIBLE INDIVIDUAL G. L. Anderson
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641
PRINCIPAL INVESTIGATOR: M. V. Gandhi and B. S. Thompson
Dept. of Mechanical Engineering
Michigan State University
Ease Lansing MI 48824-1226

OBJECTIVE: Develop a new generation of high performance light weight robot arms fabricated in advanced composite materials.

TITLE: The effects of Curvature and Thickness on Impact Damage in Cylindrical Composite Shells
RESPONSIBLE INDIVIDUAL G. L. Anderson
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641
PRINCIPAL INVESTIGATOR: F. K. Chiang
Dept. of Aeronautics and Astronautics
Standford University
Standford CA 94035-6060

OBJECTIVE: Determine the effects of curvature and thickness of the structures on impact damage in composites. Special attention is focussed on the effects of geometries and materials on the damage mechanisms and on the extent of damage in such structures. A combined analytical and experimental approach is followed.

TITLE: Stability of Elastically Tailored Rotor Systems
RESPONSIBLE INDIVIDUAL G. L. Anderson
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641
PRINCIPAL INVESTIGATOR: D. Hodges and L. Rehfield
School of Aerospace Engineering
Georgia Institute of Technology
Atlanta, GA 30332

OBJECTIVE: Develop mathematical modeling and analysis procedures to determine the aeroelastic stability characteristics of bearingless helicopter rotors on elastic supports in axial flow and tilt rotor aircraft with elastic wings in axial flight in the helicopter mode and in the airplane mode. The rotor systems are composed of or contain significant structural components fabricated from composite materials.

TITLE: Damage Resistance in Rotorcraft Structures
RESPONSIBLE INDIVIDUAL G. L. Anderson
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641
PRINCIPAL INVESTIGATOR: E. A. Armanios
School of Aerospace Engineering
Georgia Institute of Technology
Atlanta, GA 30332

OBJECTIVE: Explore the benefits of tailoring microstructure, i.e., ply stacking sequence, fiber orientation, and a blend of material plies, to contain and resist damage in rotor systems and airframe structural components. The analysis will be developed for a generic damaged ply model that includes matrix micro-cracking, delamination and fiber fracture, and their interaction.

TITLE: Optimization of Composite Drive Shafts
RESPONSIBLE INDIVIDUAL G. L. Anderson
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641
PRINCIPAL INVESTIGATOR: M. Darlow and O. A. Bachau
Dept. of Mechanical Engineering
Renssalaer Polytechnic Institute
Troy NY 12180-3590

OBJECTIVE: Develop a computerized design process for designing composite drive shafts for rotorcraft that can operate at super-critical rotational speeds. Develop algorithms to optimize shaft systems based on geometric envelope, torsional strength and elastic stability (buckling), torsional and lateral vibrations, and weight.

TITLE: Analysis and Design of Composite Fuselage Frames
RESPONSIBLE INDIVIDUAL G. L. Anderson
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641
PRINCIPAL INVESTIGATOR: O. A. Bachau
Dept. of Mechanical Engineering
Renssalaer Polytechnic Institute
Troy NY 12180-3590

OBJECTIVE: Develop a model that will allow the accurate analysis and design of helicopter fuselage frame components using composite materials. The features to be included are strong curvature (height to radius of curvature ratio of the order of 1 to 3), major secondary stresses (crushing and curling) due to this curvature, sharp changes in gage thickness, and material anisotropy effects including continuously varying directions of principal axes of orthotropy.

TITLE: Hygrothermal Effects on the Elastic Properties of Tailored Composite Blades

RESPONSIBLE INDIVIDUAL: G. L. Anderson
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641

PRINCIPAL INVESTIGATOR: S. J. Winckler
Dept. of Mechanical Engineering
Rensselaer Polytechnic Institute
Troy, NY 12180-3590

OBJECTIVE: Develop analytical models for predicting changes in stiffness and coupling during hygrothermal conditioning. Perform experiments to measure such effects.

TITLE: Formal Optimization Procedures for Composite Blades

RESPONSIBLE INDIVIDUAL: G. L. Anderson
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641

PRINCIPAL INVESTIGATOR: O. A. Bauchau
Dept. of Mechanical Engineering
Rensselaer Polytechnic Institute
Troy, NY 12180-3590

OBJECTIVE: For the full application of composite materials to rotor blades, develop design and optimization tools that allow for the imposition of multiple constraints.

TITLE: Advanced Composite Laminates for Rotorcraft

RESPONSIBLE INDIVIDUAL: G. L. Anderson
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641

PRINCIPAL INVESTIGATOR: R. J. Diefendorff, O. A. Bauchau and S. J. Winkler
Dept. of Mechanical Engineering
Rensselaer Polytechnic Institute
Troy, NY 12180-3590

OBJECTIVE: Analytical modeling, fabrication and testing research will be undertaken to develop new two and three dimensional composite concepts that promise advanced elastic tailoring, improved load transfer and/or reduced fabrication costs. Analysis methodology will be developed that is capable of predicting the elastic characteristics of laminates with "bend" and "splay" intralaminate fiber concepts.

TITLE: Finite Element Modelling of Composite Rotor Blades

RESPONSIBLE INDIVIDUAL: G. L. Anderson
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641

PRINCIPAL INVESTIGATOR: S. Lee and A. Vizzini
Dept. of Aerospace Engineering
University of Maryland
College Park, MD 20742

OBJECTIVE: Develop a beam finite element formulation of the combined dynamic bending, torsional, and extensional behavior of composite rotor blades taking into account the warping effect of blades undergoing large deflection or finite rotation. This new approach models thin walled composite blades with complicated cross sections, tapers, and arbitrary planforms. The warping effect is incorporated by assuming warping displacements superimposed over cross sections normal to the beam axis in the deformed configuration of a shear-flexible beam. Numerical tests of simple static problems demonstrate the validity and effectiveness of this approach.

TITLE: Vibration Control and Optimization in Composite Structural elements by Use of Add-on Damping Materials
RESPONSIBLE INDIVIDUAL: G. L. Anderson
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641
PRINCIPAL INVESTIGATOR: C. T. Sun and P. Hajela
Dept. of Aerospace Engineering
University of Florida
Gainesville, FL 32611
OBJECTIVE: Develop techniques of vibration control through add-on damping for structures fabricated from composite materials. Apply optimization techniques for the purpose of determining the optimum structural/damping treatment design.

TITLE: Structural Characterization and Mechanical Behavior of Chopped Fiber and Molecular Composites
RESPONSIBLE INDIVIDUAL: I. Ahmad
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641
PRINCIPAL INVESTIGATOR: James White
University of Akron OHIO

OBJECTIVE: Characterize the orientation and correlate with the mechanical behavior of (1) chopped fiber and (2) molecular composites. Both composites will be based on poly-p-phenylene terephthalamide (PPD-T).

TITLE: On the Dynamic Behavior of 3-D Integrated Fabric-Reinforced Composites **RESPONSIBLE INDIVIDUAL:** I. Ahmad
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641
PRINCIPAL INVESTIGATOR: F. K. Ko and H. Rogers
Drexel University
Philadelphia, PA 19104

OBJECTIVE: To investigate the dynamic impact resistance of 3-D braided composites, using drop weight impact test and compressed air gun techniques.

TITLE: Manufacturing Science, Reliability and Maintainability Technology **RESPONSIBLE INDIVIDUAL:** A. Crowson
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641
PRINCIPAL INVESTIGATORS: T. W. Chou and R. L. McCullough
Center For Composite Materials
University of Delaware
Newark DE 19716

OBJECTIVE: This University Research Initiative Program consists of the following elements : cure characterization and monitoring, on-line intelligent non destructive evaluation for in-process control of manufacturing, process simulation, computer aided manufacturing by filament winding, structural property relationships, mechanics of thick section composite laminates, structure performance and durability and integrated engineering for durable structures.

TITLE: Fiber Reinforced Structures
RESPONSIBLE INDIVIDUAL: J. Wu
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641
PRINCIPAL INVESTIGATORS: W. G. Strang
Dept. of Mathematics
Massachusetts Institute of Technology
Cambridge MA 02139

OBJECTIVE: Methods for optimal design of structures with special emphasis on the optimal placement of fibers to achieve maximum specific strength. The approach will be based on application of convex analysis to variational principles with inequality constraints.

TITLE: Optimal Control and Design of Composites and Layered Structures
RESPONSIBLE INDIVIDUAL: J. Wu
Army Research Office
P. O. Box 12111
Research Triangle Park, NC 27709-2211
(919) 549-0641

PRINCIPAL INVESTIGATORS: W. A. Hager
Pennsylvania State University
University Park, PA 16802

OBJECTIVE: To investigate composite materials with periodic structures, elastic wave propagation in stratified media and design materials with desired properties.

U. S. ARMY LABORATORY COMMAND
MATERIALS TECHNOLOGY LABORATORY

TITLE: Composite Hull for Light Infantry Fighting Vehicle
PROJECT ENGINEER: W. Haskell
U.S. Army Laboratory Command
Materials Technology Laboratory
Watertown, MA 02172-0001
(617) 923-5172
PRINCIPAL INVESTIGATOR: E. Weerth
FMC Corp.
San Jose, CA

OBJECTIVE: Demonstrate the application of thick composites technology to armored vehicles for the purpose of weight reduction. Payoffs in the form of reduced weight over aluminum for equal ballistic protection, reduced spall, elimination of corrosion, signature reduction, reduced life cycle costs and logistic improvements related to easier transportability and lowered fuel consumption, are being demonstrated.

TITLE: Lightweight Towbar for Battlefield Recovery
PROJECT ENGINEER: G. Piper
U.S. Army Laboratory Command
Materials Technology Laboratory
Watertown, MA 02172-0001
(617) 923-5745
PRINCIPAL INVESTIGATOR: G. Samavedam
Foster-Miller Inc.
350 Second Ave.
Waltham, MA 02254

OBJECTIVE: The objective of this effort is demonstration of the ability of composites to replace steel for weight reduction in tow bars used for battlefield recovery of the M1 tank and similar heavy vehicles. Scope includes design, fabrication and demonstration of a selected towbar design. Combinations of glass and graphite reinforced materials using filament winding and braiding manufacturing approaches are under study.
Battlefield ruggedness of the composite towbar is an overriding consideration in judging the success of the program.

TITLE: Design Analysis of Composite Test Specimens
PROJECT ENGINEER: D. W. Oplinger
U.S. Army Laboratory Command
Materials Technology Laboratory
Watertown, MA 02172-0001
(617) 923-5303
PRINCIPAL INVESTIGATOR: S. Chaterjee
Materials Sciences Corp.
Gwynedd Plaza II
Bethlehem Pike
Spring House PA
(215) 542-8400

OBJECTIVE: The objective is to evaluate current specimen designs for mechanical-property test specimens for composites and to develop design improvements. Effort includes combined stress analysis effort to evaluate specimen designs of interest, and experimental effort to provide confirmatory data, both for evaluation of current specimens and assessment of suggested improvements. Mechanical tests of interest include compression, in plane shear, interlaminar shear and tension.

TITLE: Lightweight Howitzer Project

RESPONSIBLE INDIVIDUAL: D. Tracey
U.S. Army Laboratory Command
Materials Technology Laboratory
Watertown, MA 02172-0001
(617) 923-5427

PRINCIPAL INVESTIGATOR: D. W. Oplinger
U.S. Army Laboratory Command
Materials Technology Laboratory
Watertown, MA 02172-0001
(617) 923-5303

OBJECTIVE: The objective of this effort is to demonstrate the application of composite materials to towed artillery, for the purpose of weight reduction. Available air transport facilities for many of the Army's operating units do not allow the transport of weapons of sufficient effectiveness because of the weight of such weapons. Scope of the effort includes design, fabrication and demonstration of various howitzer components to provide confidence in the ability of such materials to provide battlefield ruggedness.

TITLE: Failure and Degradation of Adhesive Joints

RESPONSIBLE INDIVIDUAL: D. Tracey
U.S. Army Laboratory Command
Materials Technology Laboratory
Watertown, MA 02172-0001
(617) 923-5427

PRINCIPAL INVESTIGATOR: R. Barsoum
U.S. Army Laboratory Command
Materials Technology Laboratory
Watertown, MA 02172-0001
(617) 923-5259

OBJECTIVE: The objective is to provide the Army with improved design, life prediction and reliability methodology for adhesive joints. The effort includes analytical and experimental efforts aimed at investigating:(1) development of improved analytical methods for fracture of adhesive joints, including adhesive bond failures as well as cohesive failures in composite adherends;(2) application of moire interferometry to evaluating pre-cracked bending specimens for adhesive testing;(3) provision of an up-to-date assessment of joint stress analysis and design methodology; (4) development of methodology for 3-D modelling of joints;(5) investigation of environmental degradation effects;(6) development of improved finite element approaches for modelling adhesive joints.

TITLE: Computational Mechanics of Thick Composites

RESPONSIBLE INDIVIDUAL: D. Tracey
U.S. Army Laboratory Command
Materials Technology Laboratory
Watertown, MA 02172-0001
(617) 923-5427

PRINCIPAL INVESTIGATOR: A. Tessler
U.S. Army Laboratory Command
Materials Technology Laboratory
Watertown, MA 02172-0001
(617) 923-5356

OBJECTIVE: The program objective is to develop analytic methods for predicting mechanical response and failure of advanced thick composite structures which may involve stress concentrators such as cutouts, fasteners, delaminations and defects. The analytic development is facilitated by a comprehensive experimental verification involving modal analysis and moire methods. The program approach encompasses: (1) development of an effective and yet simple higher-order laminated composite shell theory which would be amenable to finite element modeling to simulate linear and nonlinear dynamic response; (2) development of reliable and efficient finite element models for the analysis of composite shell structures and adhesively bonded composite joints; (3) experimental validation of the analytic and computational models via modal analysis and moire strain methods. The technology will improve the design methods for Army's thick composite structures such as those employed in helicopter rotor blades, tank hulls and turrets, light-weight howitzers, kinetic energy projectiles, and a whole range of other applications.

TITLE: Strain Concentration Assessment of Laminates Subjected to Low Velocity Impact

RESPONSIBLE INDIVIDUAL: D. Tracey
U.S. Army Laboratory Command
Materials Technology Laboratory
Watertown, MA 02172-0001
(617) 923-5427

PRINCIPAL INVESTIGATOR: S. Serabian, R. M. Anastasi
U.S. Army Laboratory Command
Materials Technology Laboratory
Watertown, MA 02172-0001
(617) 923-5260

OBJECTIVE: The objective of this effort is to investigate the effects of low velocity impact upon the structural integrity of composite laminates. A correlation between impact energy, damage zone size, and strain concentration (resulting from service loads) will be experimentally attempted for three classes of fiberglass laminates.

TITLE: Experimental Displacement Contouring of Pin Loaded Plates

RESPONSIBLE INDIVIDUAL: D. Tracey
U.S. Army Laboratory Command
Materials Technology Laboratory
Watertown, MA 02172-0001
(617) 923-5427 **PRINCIPAL INVESTIGATOR:** S. Serabian
U.S. Army Laboratory Command
Materials Technology Laboratory
Watertown, MA 02172-0001
(617) 923-5260

OBJECTIVE: The objective of this work unit is to apply conventional moire methodologies to pinloaded laminates to obtain both inplane and out of plane displacement components. A displacement contouring load history of fiberglass epoxy 0/90, +45/-45, and 0/90/+45/-45 laminates will be undertaken. Recent advances in data analysis will be employed to obtain full field strain surfaces from these experimental displacement contours thus providing an experimental data base for comparisons with on-going 3D finite element modeling activities. The effects of laminate orientation upon mechanical response will be investigated.

TITLE: Three Dimensional Finite Element Modelling of Pin Loaded Laminates

RESPONSIBLE INDIVIDUAL: D. Tracey
U.S. Army Laboratory Command
Materials Technology Laboratory
Watertown, MA 02172-0001
(617) 923-5427

PRINCIPAL INVESTIGATOR: S. Serabian
U.S. Army Laboratory Command
Materials Technology Laboratory
Watertown, MA 02172-0001
(617) 923-5260

OBJECTIVE: The objective of this work unit is to model 0/90, +45/-45, and 0/90/+45/-45 pinloaded laminates and obtain 3D finite element approximations to their mechanical response. Both linear elastic and nonlinear elastic approximations will be undertaken. Three dimensional constitutive equations will be derived from a 2D lamina characterization of the fiber-resin system and application of laminate theory. Laminate tensile and 3 point bending tests will be conducted to obtain transverse shear and through thickness Poison ratio properties. Comparisons to ongoing experimental displacement contouring activities will highlights the effects of material linearity modeling assumptions.

TITLE: Dynamics of Structures
RESPONSIBLE INDIVIDUAL: R. Shufford
U.S. Army Materials Technology Laboratory
ATTN: SLCMT-MEC
Watertown, MA 02172
(617) 923-5572
PRINCIPAL INVESTIGATORS: G.E. Foley, J. McMorrow, M.E. Roylance
U.S. Army Materials Technology Laboratory
ATTN: SLCMT-MEC
Watertown, MA 02172
(617) 923-5514

OBJECTIVE: The objective of this program is the development of modal analysis as a predictive technique for detection of damage in composite structures, such as foam core sandwich panels. This is done by characterizing the structure in terms of its modal parameters. Changes in the damping, natural frequencies, as well as the mode shapes are investigated as a function of damage in composite structures.

TITLE: Effect of Fabrication Variables on Composite Structures
RESPONSIBLE INDIVIDUAL: R. Shufford
U.S. Army Materials Technology Laboratory
ATTN: SLCMT-MEC
Watertown, MA 02172
(617) 923-5572
PRINCIPAL INVESTIGATORS: G.E. Foley, S. Ghiorse, M.E. Roylance
U.S. Army Materials Technology Laboratory
ATTN: SLCMT-MEC
Watertown, MA 02172
(617) 923-5514

OBJECTIVE: The objective of this program is to determine the advantages and/or disadvantages of different manufacturing techniques, such as braiding, filament winding, and hand lay-up on the mechanical properties, as well as the environmental durability of these composites.

TITLE: Automated Evaluation of Composite Materials
RESPONSIBLE INDIVIDUAL: G.L. Hagnauer
U.S. Army Materials Technology Laboratory
ATTN: SLCMT-EMP
Watertown, MA 02172-0001
PRINCIPAL INVESTIGATORS: G.L. Hagnauer and S.G.W. Dunn
Polymer Research Branch
U.S. Army Materials Technology Laboratory

OBJECTIVES: The objectives of this project are to increase laboratory productivity and improve the quality of test information needed to evaluate fiber-reinforced polymeric matrix composite materials and guide their specification, design and manufacture. Laboratory robotics and artificial intelligence (AI) technologies are being developed to meet requirements for handling and testing large numbers of specimens under a wide range of conditions and to increase the efficiency and reduce labor costs involved in evaluating composite materials. To control automation and deal with the large amounts of information generated by automated testing, advanced computer and AI technologies (e.g., expert systems, image analysis and machine learning) are being implemented. AI techniques will be employed to advise and plan tests, control robots and automated test equipment, and interpret and preserve information in a living database and as reports with fully traceable documentation. Currently, the technology is being implemented in research on the durability evaluation and life prediction of composites.

TITLE: Dynamic characterization of Advanced Materials
RESPONSIBLE INDIVIDUAL: J. Nunes
U.S. Army Materials Technology Laboratory
ATTN: SLCMT-MRM-MTG
Watertown, MA 02172-0001
(617) 923 5554
PRINCIPAL INVESTIGATORS: W. Crenshaw
U.S. Army Materials Technology Laboratory
ATTN: SLCMT-MRM-MTG
Watertown, MA 02172-0001
(617) 923 5203

OBJECTIVE: Design and evaluate instrumentation systems and experimental designs to measure load response of advanced composites and homogeneous materials to low speed impact loading. Conduct standard material evaluations to determine residual strength oaf the material after impact. Determine the accuracy if present constitutive models in predicting the dynamic behavior of composites and homogeneous materials during low speed impact.

TITLE: Ultrasonic Digital Signal Processing(DSP)
RESPONSIBLE INDIVIDUAL: A. Broz
U.S. Army Materials Technology Laboratory
ATTN: SLCMT-MRM
Watertown, MA 02172-0001
(617) 923 5285
PRINCIPAL INVESTIGATORS: J. Gruber
U.S. Army Materials Technology Laboratory
ATTN: SLCMT-MRM
Watertown, MA 02172-0001
(617) 923 5443

OBJECTIVE: Digital Signal Processing(DSP) is a computer based technique for enhancement of digital signals related to image processing. DSP techniques are being developed that enhance ultrasonic inspection capabilities in conjunction with NDE of composites and other applications. For example, DSP can provide additional information about individual plies in thin-lamina composites. It has also been used for the evaluation of thin bondlines in adhesive bonds. NDE efforts at MTL will emphasize the use of DSP in conjunctions with ultrasonic B scans which have been used for void location, volume fraction determination and imaging of the layered structure of laminates.

U. S. ARMY LABORATORY COMMAND
BALLISTIC RESEARCH LABORATORY

TITLE: Composite Structures

PRINCIPAL INVESTIGATOR: W.H. Drysdale
AMC LABCOM
Ballistic Research Laboratory
Aberdeen Proving Ground
MD 240055066
(301) 278-6132

OBJECTIVE: The objective of this project is to develop failure criteria, architecture transition technology, and optimum design technology for thick ballistic structures. Rate of loading and layup transition studies are being addressed at BRL. A special, high-rate, propellant driven test apparatus is under development to generate uniaxial or triaxial stress states at strain rates of up to 200 per second. Three dimensional failure criteria and other constitutive effects are being studied and hypothesized by Lawrence Livermore National Lab(LLNL). They are also sponsoring studies at the University of Utah and Pennsylvania State University. Experimental activities to develop failure data are being conducted at both the LLNL and the University of Utah. Additional failure criteria work and extensions to optimal notions for relatively simple structures and layup stacking sequences are under investigation at the AFWAML. This work also features biaxial failure tests. Initial efforts have been focussed on a reliable 3-D failure criterion for thick sections under multiaxial state of stress. Experimental data is being generated for evaluation of separate formulations. Rate, fatigue, and repair issues will be investigated next.

U. S. ARMY MISSILE COMMAND

TITLE: Determination of Mechanical Material Properties for Filament Wound Structures
RESPONSIBLE INDIVIDUAL: Dr. Larry C. Mixon
Army Missile Command
PRINCIPAL INVESTIGATOR: Terry L. Vandiver
Army Missile Command
(205) 876-1015

OBJECTIVE: The objective of this task is to develop test standards for the determination of mechanical material properties for filament wound composite structures. The initial task is to develop uniaxial material properties. Future plans include biaxial and triaxial material property determination. This effort is being performed by the Joint-Army-Navy-NASA-Air Force (JANNAF) Composite Motor Case Subcommittee through a round robin test effort. This task is coordinated with MIL-HDBK-17, ASTM, National Bureau of Standards, and DoD CMPS Composites Technology Program.

TITLE: Composite Materials Evaluation for Filament Winding
RESPONSIBLE INDIVIDUAL: Lawrence W. Howard
Army Missile Command
PRINCIPAL INVESTIGATOR: Terry L. Vandiver
Army Missile Command
(205) 876-1015

OBJECTIVE: The object of this task is to evaluate new fibers for filament winding. Delivered strengths are determined via strand tests and 3-inch diameter filament wound pressure vessels with different stress ratios. The experimental data is used in the design of composite rocket motor cases, launchers, pressure vessels and other filament wound structures.

TITLE: Composite Wing Design and Fabrication
RESPONSIBLE INDIVIDUAL: Lawrence W. Howard
Army Missile Command
PRINCIPAL INVESTIGATORS: J. Frank Wlodarski
Terry L. Vandiver
Army Missile Command
(205) 876-0398

OBJECTIVE: The objective of this task is to design and fabricate an all composite wing with an elliptical planform. The materials used are s-glass cloth and uni-directional tape. These materials were selected because of their strength, stiffness and low radar cross-section. The method of fabrication is hand layup in a clamshell mold made of composite tooling. The wings are tested to determine what structural properties are achieved with this method of manufacture and if they are accurately predicted in the design.

TITLE: Buckling of Composite Cylinders
RESPONSIBLE INDIVIDUAL: Lawrence W. Howard
Army Missile Command
PRINCIPAL INVESTIGATOR: J. Frank Wlodarski
Army Missile Command
(205) 876-0398

OBJECTIVE: To determine and improve the buckling strength of a composite cylinder subjected to external pressure. Several cylinders 1.754 in diameter were filament wound with different layups and tested in a hydrostatic test fixture to measure the buckling strength. From the results of the hydrostatic test the predicted strength can be compared with the experimental strength and improvements can be made by changing the layup.

TITLE: Evaluation of Finite Element Codes
RESPONSIBLE INDIVIDUAL: Lawrence W. Howard
Army Missile Command
PRINCIPAL INVESTIGATORS: David McNeill and J. Frank Wlodarski
Army Missile Command
(205) 876-0398

OBJECTIVE: The principal objectives of this evaluation were to find a commercially available PC based finite element code that can be utilized with little training. The main specifications that the codes were evaluated against were user friendliness of the pre and post processors, clarity of user manuals, accuracy of results and ability to handle composite structures.

U. S. ARMY AVIATIONS SYSTEMS COMMAND
FT. EUSTIS

TITLE: Damage Tolerance Testing of the ACAP Roof
RESPONSIBLE INDIVIDUAL: F. Swats
U.S. Army ARTA (AVSCOM)
Aviation Applied Technology Directorate
SAVRT-TY-ATS
Ft. Eustis, VA 23604-5577
(804) 878-2975
PRINCIPAL INVESTIGATOR: B. Spigel

OBJECTIVE: A forward roof subcomponent from the Bell Advanced Composite Airframe Program (ACAP) helicopter will be tested to verify the damage tolerance design criteria developed under contract by Bell Helicopter Textron, Inc. (Final Report: USAAVSCOM TR-87-D-3A, B, C). The roof will be subjected to an anticipated ACAP load spectrum, and manufacturing defects and in-service damage will be monitored by both laboratory and field nondestructive evaluation methods to determine the extent of damage growth.

TITLE: Ballistic Survivability of Generic Composite Main Rotor Hub Flexbeams
RESPONSIBLE INDIVIDUAL: F. Swats
U.S. Army ARTA (AVSCOM)
Aviation Applied Technology Directorate
SAVRT-TY-ATS
Ft. Eustis, VA 23604-5577
(804) 878-2975
PRINCIPAL INVESTIGATORS: E. Robeson and K. Sisitka

OBJECTIVE: The goal of this effort is to quantify the ballistic survivability of typical composite main rotor hub flexbeams. Two different flexbeam designs will be impacted with various ballistic threats. One design will be tested under simulated centrifugal load while the other will be fatigue tested following ballistic impact in a no load condition. Fatigue testing of the first design will be considered after a damage assessment is made.

TITLE: Finite Element Correlation of the Advanced Composite Airframe Program (ACAP) Dynamic Models
RESPONSIBLE INDIVIDUAL: E. Austin
U.S. Army ARTA (AVSCOM)
Aviation Applied Technology Directorate
SAVRT-TY-ATS
Ft. Eustis, VA 23604-5577
(804) 878-3822
PRINCIPAL INVESTIGATORS: N. Calapodas and K. Hoff
Bell Helicopter TExtron
P.O. Box 482
Ft. Worth, TX 76101

Sikorsky Aircraft
North Main Street
Stratford, CT 06601

OBJECTIVE: A joint program among Army/NASA/Contractor is planned to conduct detail correlation of the Finite Element (FE) dynamic models of both ACAP airframes. AATD will perform all shake testing and the contractors will be responsible for analytical changes to the FE models. The FE dynamic models, generated under Army funding during the develop-mental phase of the ACAP program, were further improved under funding of the NASA DAMVIBS program. However, the thrust of shake testing performed during the developmental phase was oriented towards qualifying the flightworthy vehicles for flight testing rather than conducting detail correlation of the FE models. The results obtained from the limited shake testing identified substantial discrepancies between test and analysis, limiting the usefulness of the models to 15 Hz and below. In the correlation to be performed, the test vehicles will be stripped down to the basic structure. The inertia of the components removed will be substituted with

concentrated masses. Upon successful correlation of the basic configuration, components will be installed and correlation efforts repeated. The goal is to achieve satisfactory correlation at modal and force response frequencies up to 40 Hz.

U. S. ARMY AVIATIONS SYSTEMS COMMAND
US ARMY RESEARCH & TECHNOLOGY ACTIVITY
NASA-LANGLEY RESEARCH CENTER

TITLE: Basic Research in Structures

RESPONSIBLE INDIVIDUAL: Dr. F. D. Bartlett, Jr.
U.S. Army ARTA
Aerostructures Directorate
Mail Stop 266
NASA Langley Research Center
Hampton, VA 23665-5225
PRINCIPAL INVESTIGATORS: Dr. T.K. O'Brien, Dr. R.L. Boitnott, G.L. Farley, M.W. Nixon

OBJECTIVE: The objectives and scope of this research are to investigate and explore structures technologies which exploit advanced materials for improved structural performance, develop superior analyses for composites design, and devise automated processes for inspecting and manufacturing rotorcraft structures. This is accomplished, in conjunction with NASA Langley, by conducting basic research of composite and metallic materials to understand and improve fatigue resistance, fracture toughness, crashworthiness, and internal noise transmission as well as to develop more efficient and damage tolerant structural forms for rotorcraft applications.

TITLE: Structures Technology Applications

RESPONSIBLE INDIVIDUAL: Dr. F.D. Bartlett, Jr.
U.S. Army ARTA
Aerostructures Directorate
Mail Stop 266
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2866
PRINCIPAL INVESTIGATORS: Dr. R.L. Boitnott, M.W. Nixon, G.L. Farley, D.J. Baker

OBJECTIVE: The goals of this research are to explore and demonstrate innovative structural concepts and design methodologies which will provide lighter, safer, and more survivable structures for rotorcraft. This is achieved through jointly-sponsored Army/NASA investigations which establish improved structural integrity and crashworthiness, validate superior analytical capabilities, and demonstrate lower cost manufacturing processes. The emphasis of this research is to provide proven technology to the rotorcraft industry and the U.S. Army for applications to future air vehicle systems.

NASA LANGLEY RESEARCH CENTER

IN-HOUSE

EXPERIMENTAL CHARACTERIZATION OF THE FRACTURE BEHAVIOR OF METAL MATRIX COMPOSITES
87 June - 89 September 30

Project Engineer: Dr. W. Steven Johnson
Mail Stop 188E
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2715 FTS 928-2715

OBJECTIVE: To experimentally investigate the fatigue, fracture, and thermomechanical behavior of MMC's to insure airframe structural integrity at elevated temperatures. Both continuously reinforced laminates and discontinuous particulate and whisker reinforced MMC's will be included in the study.

DEVELOPMENT OF ANALYTICAL MODELS OF THE THERMOMECHANICAL BEHAVIOR OF METAL MATRIX COMPOSITES
87 June - 89 September 30

Project Engineer: Dr. Cathy A. Bigelow
Mail Stop 188E
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3047 FTS 928-3047

OBJECTIVE: To develop finite-element codes, laminate-analysis codes, and micromechanics models necessary to analytically investigate mechanics issues related to the fatigue, fracture, and thermomechanical behavior of MMC's.

DELAMINATION MICROMECHANICS ANALYSIS
85 October 1 - 89 September 30

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, VA 23365-5225
(804) 865-3048 FTS 928-3048

Objective: To develop a fiber-resin stress analysis for region near a delamination front subjected to Mode I loading.

EFFECT OF IMPACT ON FWC FOR SPACE SHUTTLE'S SRBs
83 August - 89 September 30

Project Engineer: Mr. C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2338 FTS 928-2338

Objective: To determine the strength loss of filament wound cases (FWCs) due to low velocity impact.

PREDICTION OF INSTABILITY-RELATED DELAMINATION GROWTH
79 January 2 - 89 September 30

Project Engineer: Dr. John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3046 FTS 928-3046

Objective: To predict the rate of instability-related delamination growth. Rigorous and approximate techniques for calculating strain-energy release rates have been developed for two-dimensional configurations. The current effort is concentrating on three-dimensional configurations. Experiments are planned for evaluation of the analytical methodology.

MECHANICS MODELS OF 3-D ADVANCED COMPOSITE FORMS
88 June 1 - 89 September 30

Project Engineer: Dr. John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3046 FTS 928-3046

Objective: To develop mechanics-based models of the deformation and local stress states that reflect the local fiber curvature of 3-D advanced composite forms. These models will form the basis for establishing strength (failure) criteria and will provide insight into an optimized material form from the mechanics viewpoint. Experiments will be conducted to support the model development and to verify predictions.

CHARACTERIZATION OF MODE I AND MODE II DELAMINATION GROWTH AND THRESHOLDS IN GRAPHITE/PEEK COMPOSITES
87 June - 89 September 30

Project Engineer: Ms. Gretchen B. Murri
Aerostructures Directorate, USAARTA (AVSCOM)
Mail Stop 188E
NASA Langley Research Center
Hampton, VA 23365-5225
(804) 865-2093 FTS 928-2093

Objective: To characterize the Mode I and Mode II delamination failures of graphite/PEEK composite using the double cantilevered beam (DCB) and end-notched flexure (ENF) tests. A fatigue delamination growth criteria and threshold value for no-crack growth will be established.

INTERLAMINAR SHEAR FRACTURE TOUGHNESS
87 May - 89 September 30
Project Engineer: Ms. Gretchen B. Murri
Mail Stop 188E
Aerostructures Directorate, USAARTA (AVSCOM)
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2093 FTS 928-2093

Objective: This program is part of a round robin of fracture toughness tests organized by ASTM Committee D30 on High Modulus Fibers and Their Composites. The end-notched flexure tests will be used to measure the Mode II strain energy release rate of two composite materials. Data reduction methods will be compared. Results will be compared with those of other test labs participating and will be used to develop ASTM test standards for interlaminar shear fracture toughness.

DELAMINATION GROWTH IN TAPERED COMPOSITE LAMINATES WITH INTERNAL PLY DROPS
86 June - 89 September 30

Project Engineer: Dr. T. Kevin O'Brien
Mail Stop 188E
Aerostructures Directorate, USAARTA (AVSCOM)
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2093 FTS 928-2093

Objective: In tapered composites containing internal ply drops which undergo tension and bending loads, delamination failures are typically observed at the locations of the ply drops. The objective of this program is to develop analyses which accurately model this delamination failure mode.

INTERLAMINAR FRACTURE TOUGHNESS TESTING OF COMPOSITES
86 April - 89 September 30

Project Engineer: Dr. T. Kevin O'Brien
Mail Stop 188E
Aerostructures Directorate, USAARTA (AVSCOM)
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2093 FTS 928-2093

Objective: In order to develop standard tests for measuring interlaminar fracture toughness of composites, ASTM Committee D30 on High Modulus Fibers and Their Composites has organized a round robin series of four test methods. The double cantilevered beam (DCB), edge delamination tension (EDT), cracked lap shear (CLS) and end-notched flexure (ENF) tests will be evaluated by a total of 32 laboratories using 3 different materials, ranging from very brittle to very tough.

EXPERIMENTAL EVALUATION OF ADVANCED COMPOSITE MATERIAL FORMS
84 June 1 - 89 June 1

Project Engineer: Mr. H. Benson Dexter
Mail Stop 188B
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2869 FTS 928-2869

Objective: To determine mechanical properties and establish damage tolerance of 2-D and 3-D woven, stitched, and braided composite materials.

FLIGHT SERVICE EVALUATION OF COMPOSITE COMPONENTS ON COMMERCIAL AND MILITARY AIRCRAFT
72 March 1 - 90 December 31

Project Engineer: Mr. H. Benson Dexter
Mail Stop 188B
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2869 FTS 928-2869

Objective: To evaluate the long-term durability of composite components installed on commercial and military transport and helicopter aircraft. Over 300 components constructed of boron, graphite, and Kevlar composites will be evaluated after extended service. Components include graphite/epoxy rudders, spoilers, tail rotors, vertical stabilizers, Kevlar/epoxy fairings, doors and ramp skins, and boron/aluminum aft pylon skins. Note: Over 4.5 million total component flight hours have been accumulated since initiation of flight service in 1972. Composite components on L-1011, B-737, and DC-10 aircraft have accumulated over 40,000 flight hours each. Excellent in-service performance and maintenance experience has been achieved with the composite components.

THE ENERGY ABSORPTION OF COMPOSITES
80 August 1 - 89 Jan 30

Project Engineer: Mr. Gary L. Farley
Mail Stop 188B
Aerostructures Directorate, USAARTA (AVSCOM)
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2850 FTS 928-2850

Objective: To develop an understanding of the energy absorption mechanisms of composite materials and how the constitutive properties and specimen architecture effect energy absorption capability. Develop subfloor structural concepts and the analytical ability to predict their energy absorption.

ADVANCED CONCEPTS FOR COMPOSITE HELICOPTER FUSELAGE STRUCTURES
83 April 1 - 92 January 1

Project Engineer: Mr. Donald J. Baker
Mail Stop 188B
Aerostructures Directorate, USAARTA (AVSCOM)
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2850 FTS 928-2850

Objective: To investigate new design concepts for composite materials on lightly loaded helicopter fuselage structures. Trade studies will be performed using the computer code PASCO. A 4-year task assignment contract will be awarded in Fiscal Year 1988 to fabricate selected designs that will be tested at NASA Langley.

MICROMECHANICS MODELING OF COMPOSITE THERMOELASTIC BEHAVIOR
86 October 1 - 89 June 30

Project Engineer: Mr. David E. Bowles
Mail Stop 188B
NASA Langley Research Center
Hampton, VA 23665-5225
(804)865-4558 FTS 928-4558

Objective: Develop analytical methods to investigate thermally induced deformations and stresses in continuous fiber reinforced composites at the micromechanics level, and predict how these deformations and stresses affect the dimensional stability of the composite.

ADVANCED CONCEPTS FOR COMPOSITE HELICOPTER FUSELAGE STRUCTURES
83 April 1 - 92 January 1

Project Engineer: Mr. Donald J. Baker
Mail Stop 188B
Aerostructures Directorate, USAARTA (AVSCOM)
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2850 FTS 928-2850

Objective: To investigate new design concepts for composite materials on lightly loaded helicopter fuselage structures. Trade studies will be performed using the computer code PASCO. A 4-year task assignment contract will be awarded in Fiscal Year 1988 to fabricate selected designs that will be tested at NASA Langley.

POSTBUCKLING AND CRIPPLING OF COMPRESSION-LOADED COMPOSITE STRUCTURAL COMPONENTS
79 March 1 - 89 September 30

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2552 FTS 928-2552

Objective: To study the postbuckling and crippling of compression-loaded composite components and to determine the limitations of postbuckling design concepts in structural applications.

DESIGN TECHNOLOGY FOR STIFFENED CURVED COMPOSITE PANELS AND SHELLS
79 October 1 - 89 September 30

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2552 FTS 928-2552

Objective: To develop verified design technology for generic advanced-composite stiffened curved panels.

POSTBUCKLING OF FLAT STIFFENED GRAPHITE/EPOXY SHEAR WEBS
81 July 1 - 89 September 30

Project Engineer: Mr. Marshall Rouse
Mail Stop 190
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-4585 FTS 928-4585

Objective: To study the postbuckling response and failure characteristics of flat stiffened graphite/epoxy shear webs.

POSTBUCKLING ANALYSIS OF GRAPHITE/EPOXY LAMINATES
80 October 1 - 89 September 30

Project Engineer: Dr. Manuel Stein
Mail Stop 190
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2813 FTS 928-2813

Objective: To develop accurate analyses for the postbuckling response of graphite/epoxy laminates and to determine the parameters that govern postbuckling behavior.

CRASH CHARACTERISTICS OF COMPOSITE FUSELAGE STRUCTURES
82 July 1 - 89 September 30

Project Engineer: Mr. Huey D. Carden
Mail Stop 495
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3795 FTS 928-3795

Objective: To study the crash characteristics of composite transport fuselage structural components.

BUCKLING AND STRENGTH OF THICK-WALLED COMPOSITE CYLINDERS
86 October 1 - 89 September 30

Project Engineer: Ms. Dawn C. Jegley
Mail Stop 190
NASA Langley Research Center
Hampton, VA 23665-5225
(804)865-4052 FTS 928-4052

Objective: To develop accurate analyses for the buckling and strength predictions of thick-walled composite cylinders.

ADVANCED COMPOSITE STRUCTURAL CONCEPTS
84 October 1 - 89 September 30

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2552 FTS 928-2552

Objective: To develop composite structural concepts and design technology needed to realize the improved performance, structural efficiency, and lower-cost advantage offered by new material systems and manufacturing methods for advanced aircraft structures.

COMPRESSION STRENGTH OF COMPOSITE LAMINATES WITH DAMAGE AND LOCAL DISCONTINUITIES
76 October 1 - 89 September 30

Project Engineer: Dr. Mark J. Shuart
Mail Stop 190
NASA Langley Research Center
Hampton, VA 23365-5225
(804) 865-2813 FTS 928-2813

Objective: To study the effects of impact damage and local discontinuities on the compression strength of composite structural components, to identify the failure modes that govern the behavior of compression loaded components subjected to low-velocity impact damage, and to analytically predict failure and structural response.

MECHANICS OF ANISOTROPIC COMPOSITE STRUCTURES
86 October 1 - 89 September 30

Project Engineer: Dr. Michael P. Nemeth
Mail Stop 190
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-4052 FTS 928-4052

Objective: To develop analytical procedures for anisotropic structural components that accurately predict the response of tailored structures.

COMPOSITES CHARACTERIZATION WORK BREAKDOWN STRUCTURE
1975 - Present

Project Engineer: Dr. Joseph S. Heyman
IRD, Nondestructive Measurement Science Branch
Mail Stop 231
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3036 FTS 928-3036

Objective: Develop quantitative measurement and technology to characterize properties of composites nondestructively and link physical properties thus measured to engineering properties needed for materials and structures certification.

QUANTITATIVE MEASUREMENTS OF MATERIAL PROPERTIES
1985 Jan - 1992 September

Project Engineer: Dr. Eric I. Madaras
IRD, Nondestructive Measurement Science Branch
Mail Stop 231
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3036 FTS 928-3036

Objective: The objective of this work is to develop advanced measurement systems for improved nondestructive characterization of composite materials. Recent work has included quantitative evaluations of porosity and other material defects in composites and the measurement of moduli in carbon-carbon composites.

IMAGE ENHANCEMENT TECHNIQUES FOR QUANTITATIVE EVALUATION STUDIES OF COMPOSITE MATERIALS
1986 June - 1990 September

Project Engineer: Dr. Patrick H. Johntson
IRD, Nondestructive Measurement Science Branch
Mail Stop 231
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3036 FTS 928-3036

Objective: To combine quantitative techniques of measurement science with methods of image production, enhancement, and display to aid in the nondestructive characterization and evaluation of composite structures.

DEVELOP NDE MEASUREMENT TECHNIQUES TO DETERMINE STATE OF FATIGUE FOR AEROSPACE MATERIALS
1988 September - 1993 September

Project Engineer: Dr. William T. Yost
IRD, Nondestructive Measurement Science Branch
Mail Stop 231
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3036 FTS 928-3036

Objective: To compare engineering-based fatigue properties with other physical measurements (including microstructural characteristics) in metals. This is to be done through cooperation of and with our Materials Division, when appropriate.

DEVELOP COMPOSITE MATERIALS WITH EMBEDDED FIBER-OPTIC SENSORS FOR IN-SITU MONITORING OF MATERIAL PROPERTIES
1988 September - 1991 September

Project Engineer: Dr. Robert S. Rogowski
IRD, Nondestructive Measurement Science Branch
Mail Stop 231
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3036 FTS 928-3036

Objective: Investigate fiber-optic sensors as potential internal sensors for composite materials. The embedded sensors are intended to monitor cure processing and subsequently serve as sensors for strain temperature, physical and chemical damage and other parameters important to the function of the material during use.

USE OF ULTRASONIC TECHNIQUES FOR COMPOSITE CURE MONITORING AND CHARACTERIZATION OF RESIN SYSTEM DURING PROCESSING
1983 September - 1990 September

Project Engineer: Dr. William P. Winfree
IRD, Nondestructive Measurement Science Branch
Mail Stop 231
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3036 FTS 928-3036

Objective: The objective of this program is to develop techniques for determining the material properties of composites during their cure. These material properties can be used as inputs to a process controller which can tailor a process to maximize the integrity of a composite. The research has concentrated on using ultrasonic techniques, with both conventional transducers and acoustic wave guides as sensors.

QUANTITATIVE ACOUSTIC EMISSION ANALYSIS OF ADHESIVE BOND FAILURE

Project Engineer: Mr. William H. Prosser
IRD, Nondestructive Measurement Science Branch
Mail Stop 231
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3036 FTS 928-3036

Objective: The objective is to study the influence of fracture toughness of the adhesive, mode of fracture, and crack velocity on the acoustic emission released during adhesive bond failure.

ANALYSIS OF COMPOSITE/HONEYCOMB PANELS FOR PRECISION REFLECTORS
88 June 1 - 90 May 31

Project Engineer: Mr. David E. Bowles
Mail Stop 188B
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-4558 FTS 928-4558

Objective: Determine the effects of constituent properties on thermally induced deformations in composite/honeycomb panels.

CONTRACTS

CRACK PROBLEMS IN ORTHOROPIC PLATES AND NONHOMOGENEOUS MATERIALS
NAG-1-713
86 November 1 - 88 October 31

Project Engineer: Dr. C. A. Bigelow
Mail Stop 188E
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3047 FTS 928-3048

Principal Investigator: Dr. Fazil Erdogan
Department of Mechanical Engineering and Mechanics
Lehigh University
Bethlehem, PA 18015
(215) 758-3000

Objective: The objective of this program is the study of plate and shell structures containing surface cracks under mixed mode loading conditions, the consideration of crack closure on the compression side of plate with a through crack under bending, the determination of the profile of a subcritically growing crack in a plate under bending and membrane loading, and the modeling of the interface region in bonded materials.

THE INFLUENCE OF WAVY LAYERS ON LAMINATE STRENGTH
NAG-1-711
88 January 15 - 89 January 15

Project Engineer: Mr. Clarence C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2338 FTS 928-2338

Principal Investigator: Dr. Alton L. Highsmith
Department of Aerospace Engineering/Texas Engineering Experiment Station
Texas A&M University System
College Station, TX 77843

Objective: Determine the influence of wavy layers on the tensile and compressive strength of laminated composites using an analysis based on variational principles. Experiments with Moire interferometry will be used to verify the analysis.

FRACTURE OF THICK COMPOSITE LAMINATES DUE TO COMBINED TENSILE AND BENDING LOADS
NAG-1-264
87 January 1 - 88 December 31

Project Engineer: Mr. Clarence C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2338 FTS 928-2338

Principal Investigators: Dr. D. H. Morris and Dr. R. A. Simonds
Department of Engineering Science and Mechanics
Virginia Polytechnic Institute and State University
Blacksburg, VA 24061

Objective: Compare linear elastic fracture mechanics with the fracture behavior of a thick, quasi-isotropic graphite/epoxy laminate with surface cuts subjected to simultaneous tensile and bending loads.

MICROMECHANICS OF COMPOSITE LAMINATE COMPRESSIVE FAILURE
NAG-1-659
86 February 1 - 89 June 30

Project Engineer: Dr. John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3046 FTS 928-3046

Principal Investigator: Dr. Walter Bradley
Department of Mechanical Engineering
Texas A&M University
College Station, TX 77843

Objective: The objective of this program is to characterize the compressive failure behavior of notched laminates under static and sustained loads. Both room temperature and elevated temperature conditions are being examined.

ANALYSIS OF DELAMINATION RELATED FRACTURE PROCESSES IN COMPOSITES
NAG-1-637
88 October 1 - 89 September 30

Project Engineer: Dr. T. Kevin O'Brien
Aerostructures Directorate, USAARTA (AVSCOM)
NASA Langley Research Center
Mail Stop 188E
Hampton, VA 23665-5225
(804) 865-2093 FTS 928-2093

Principal Investigator: Dr. E. A. Armanios
School of Aerospace Engineering
Georgia Institute of Technology
Atlanta, GA 30332

Objective: The objective of this program is to extend an existing sublaminate analysis method to model tapered ply-drop configurations and delaminations initiating from internal ply cracks. The analyses are intended for use on personal class computers.

RESIDUAL THERMAL STRESSES IN METAL MATRIX COMPOSITES
L-24457C
87 July - 89 January

Project Engineer: Dr. W. S. Johnson
Mail Stop 188E
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2715 FTS 928-2715

Principal Investigator: Dr. Yehia A. Bahei-El-Din
Department of Civil Engineering
Rensselaer Polytechnic Institute
Troy, NY 12180-3590
(518) 276-8043

Objective: This contract is to develop an analytical method for estimating residual thermal stresses in continuous fiber-reinforced metal matrix composites due to fabrication and/or subsequent thermal cycling.

FRACTOGRAPHY OF COMPOSITE MATERIALS
NAG-1-705
86 October 1 - 89 September 30

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, VA 23365-5225
(804) 865-3048 FTS 928-3048

Principal Investigator: Dr. W. D. Bascom
Department of Materials Science and Engineering
University of Utah
Salt Lake City, UT 84112

Objective: To analyze the laminate microdamage that accompanies delamination.

THERMALLY INDUCED INTERFACIAL STRESS-STRAIN BEHAVIOR IN RESIN MATRIX COMPOSITES
NAS1-18231
87 August 1 - 88 July 31

Project Engineer: Mr. David E. Bowles
Mail Stop 188B
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-4558 FTS 928-4558

Principal Investigator: Dr. B. N. Cox
Rockwell Science Center
P. O. Box 1085
Thousand Oaks, CA 91360
(805) 373-4287

Objective: Experimentally and analytically investigate the thermally induced interfacial stress-strain behavior in aerospace resin matrix composites.

EFFECTS OF STRESS CONCENTRATIONS IN COMPOSITE STRUCTURES
NSG-1483
78 January 15 - 89 January 14

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2552 FTS 928-2552

Principal Investigator: Dr. Wolfgang G. Knauss
California Institute of Technology
Pasadena, CA 91125
(213) 356-4524/4528

Objective: To study the effects of low-speed impact damage in composite structural components using high-speed motion pictures and to develop an analytical procedure for the propagation of the resulting impact damage.

ADVANCED COMPOSITE STRUCTURAL DESIGN TECHNOLOGY FOR COMMERCIAL TRANSPORT AIRCRAFT
NAS1-15949
79 September 24 - 90 March 23

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2552 FTS 928-2552

Principal Investigator: Dr. Sherrill B. Biggers
Lockheed-Georgia Company
86 South Cobb Drive
Marietta, GA 30063
(404) 494-5854

Objective: To design, analyze, fabricate, and test generic advanced-composite structural components for transport aircraft applications in order to develop verified design technology.

STRUCTURAL OPTIMIZATION FOR IMPROVED DAMAGE TOLERANCE
NAG-1-168
81 September 1 - 89 October 15

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2552 FTS 928-2552

Principal Investigator: Dr. Raphael T. Haftka
Virginia Polytechnic Institute and State University
Blacksburg, VA 24061
(703) 961-4860

Objective: To develop a structural optimization procedure for composite wing boxes that includes the influence of damage-tolerance considerations in the design process.

FAILURE ANALYSIS AND DAMAGE TOLERANCE OF COMPOSITE AIRCRAFT STRUCTURES
NAS1-17925
85 February 23 - 90 February 22

Project Engineer: Dr. Mark J. Shuart
Mail Stop 190
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2813 FTS 928-2813

Principal Investigator: Dr. Sherrill B. Biggers
Lockheed-Georgia Company
86 South Cobb Drive
Marietta, GA 30063
(404) 494-5854

Objective: To develop advanced structural concepts and to advance the analytical capability to predict composite structure failure.

ANISOTROPIC SHELL ANALYSIS
NAG-1-901
88 October 1 - 90 September 30

Project Engineer: Dr. Michael P. Nemeth
Mail Stop 190
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-4052 FTS 918-4052

Principal Investigator: Dr. Michael W. Hyer
Virginia Polytechnic Institute and State University
Blacksburg, VA 24061
(703) 961-5372

Objective: To develop accurate analyses for the response of anisotropic composite shell structures.

THICKNESS DISCONTINUITY EFFECTS
NAG-1-537
85 October 1 - 89 September 30

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2552 FTS 928-2552

Principal Investigator: Dr. Eric R. Johnson
Virginia Polytechnic Institute and State University
Blacksburg, VA 24061
(703) 961-6126

Objective: To develop verified analytical models of compression loaded laminates with thickness discontinuities and dropped plies.

QUANTITATIVE NONDESTRUCTIVE EVALUATION OF COMPOSITE MATERIALS BASED ON ULTRASONIC WAVE PROPAGATION
NSG-1-601
1981 September - 1990 September

Project Engineer: Dr. Eric I. Madaras
IRD, Nondestructive Measurement Science Branch
Mail Stop 231
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3036 FTS 928-3036

Principal Investigator: James G. Miller
Department of Physics
Washington University
St. Louis, MO 63130

Objective: The overall goal of our research program is the development and application of quantitative ultrasonic techniques to problems of nondestructive evaluation of composite materials. One goal of this work is to demonstrate the potential application of approaches based on the relationship between frequency dependent attenuation and dispersion to nondestructive evaluation of porosity. A second goal is the use of quantitative polar back-scatter and attenuation measurements to characterize material properties.

INVESTIGATION OF ACOUSTIC PROPERTIES OF COMPOSITE MATERIALS
NAG-1-431
1983 September - 1990 September

Project Engineer: Dr. Eric I. Madaras
IRD, Nondestructive Measurement Science Branch
Mail Stop 231
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3036 FTS 928-3036

Principal Investigator: Barry T. Smith
Department of Physics
Christopher Newport College
Newport News, VA 23606

Objective: The research involves an investigation of the ultrasonic acoustic properties of composite materials. The objective is to characterize the material as well as develop means of assessing any damage. Research to date has included quantitative measurement of impact damage in thin graphite/epoxy composites, evaluation of porosity and determination of fundamental ultrasonic properties to elucidate the propagation of ultrasonic waves in these materials.

FIBER WAVEGUIDE SENSORS FOR INTELLIGENT MATERIALS
NAG-1-7801
1988 September - 1995 September

Project Engineer: Dr. Robert S. Rogowski
IRD, Nondestructive Measurement Science Branch
Mail Stop 231
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-3036 FTS 928-3036

Principal Investigator: Richard O. Claus
Department of Electrical Engineering
Virginia Polytechnic Institute and State University
Blacksburg, VA 24061

Objective: Development of fiber-optic based opto-electronic sensing instrumentation for the characterization of materials and structures.

Current Research Goals:

1. Design and implementation of embedded optical-fiber sensors for the nondestructive monitoring of material cure, in service structural dynamics and material integrity.
2. Basic research of the operation of discrete fiber sensors and sensor systems.
3. Applied research on multi-parameter and distributed fiber sensor networks.
4. Improvement of such sensors via specialized fiber materials, geometries, coatings, and opto-electronic processing.

DEVELOPMENT OF ADVANCED WOVEN COMPOSITE MATERIALS AND STRUCTURAL FORMS
NAS1-18358
86 August 29 - 90 August 29

Project Engineer: Mr. H. Benson Dexter
Mail Stop 188B
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2869 FTS 928-2869

Principal Investigator: Ms. Janice Maiden
Textile Technologies, Inc.
2800 Turnpike Drive
Hatboro, PA 19040
(215) 443-5325

Objective: To develop textile technology to produce 2-D and 3-D woven preforms and structural elements with integral stiffening, multilayers, and mulidirectional reinforcement.

ANALYSIS OF 2-D REINFORCED COMPOSITES
NAS1-18000
87 March 1 - 90 March 1

Project Engineer: Mr. H. Benson Dexter
Mail Stop 188B
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2869 FTS 928-2869

Principal Investigator: Mr. Raymond L. Foye
PRC Kentron, Inc.
Hampton, VA 23665-5225
(804) 865-2850 FTS 928-2850

Objective: To develop analytical methods to understand and predict the elastic and strength response of 2-D and 3-D reinforced composite materials. Emphasis is on improved fracture toughness and impact resistance for woven, stitched, and braided material forms.

ENVIRONMENTAL EXPOSURE EFFECT ON COMPOSITE MATERIALS FOR COMMERCIAL AIRCRAFT
NAS1-15148
77 November 1 - 88 November 30

Project Engineer: Mr. H. Benson Dexter
Mail Stop 188B
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2869 FTS 928-2869

Principal Investigator: Mr. Randy Coggeshall
Boeing Commercial Airplane Company
P.O. Box 3707
Seattle, WA 98124
(206) 251-2705

Objective: To provide technology in the area of environmental effects on graphite/epoxy composite materials, including long-term performance of advanced resin-matrix composite materials in ground and flight environments.

DEVELOPMENT OF AN IN-PLANE SHEAR MATERIAL PROPERTY TEST FOR COMPOSITE LAMINATES
NCC-1-93
85 November 1 - 88 August 31

Project Engineer: Mr. Gary L. Farley
Mail Stop 188B
Aerostructures Directorate, USAARTA (AVSCOM)
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-2850 FTS 928-2850

Principal Investigator: Dr. John M. Kennedy
Department of Mechanical Engineering
Clemson University
Clemson, SC
(803) 656-5632

Objective: To develop an improved in-plane shear test method for determining the in-plane shear stiffness and strength of metallic and composite materials. Develop methods of introducing the load into the specimen in a uniform manner and experimentally validate the test method.

VISCOELASTIC RESPONSE OF COMPOSITE/HONEYCOMB PANELS FOR PRECISION REFLECTORS
NAG-1-343
88 Aug 16 - 90 July 31

Project Engineer: Mr. David E. Bowles
Mail Stop 188B
NASA Langley Research Center
Hampton, VA 23665-5225
(804) 865-4558 FTS 928-4558

Principal Investigator: Dr. M. W. Hyer
Virginia Polytechnic Institute and State University
Blacksburg, VA 24061
(703) 961-5372

Objective: Analytically and experimentally investigate the viscoelastic response of sandwich panels fabricated from composite facesheets and honeycomb cores.

OFFICE OF NAVAL RESEARCH
ARLINGTON, VA 22217-5000

NATIONAL CENTER FOR COMPOSITE MATERIALS RESEARCH
p400013f101
September 86 - September 91

Scientific Officer: Dr. Yapa D. S. Rajapakse
Office of Naval Research
Mechanics Division, Code 1132-SM
Arlington, VA 22217-5000
(202) 696-4405, Autovon 226-4405

Principal Investigator: Prof. S. S. Wang
University of Illinois
Department of Theoretical and Applied Mechanics
Urbana, IL 61801
(217) 333-1835

Objective: Under ONR-URI sponsorship, a National Center for Composite Materials Research was established to conduct a well structured, multidisciplinary research program in composites spanning the disciplines of solid mechanics, materials science, chemistry and surface physics. Initial emphasis will be on critical research issues associated with the use of thick composites for ship structures.

FAILURE OF THICK COMPOSITE LAMINATES
N00014-88-F-0044
February 88 - January 90

Scientific Officer: Dr. Yapa D. S. Rajapakse
Office of Naval Research
Mechanics Division, Code 1132-SM
Arlington, VA 22217-5000
(202) 696-4405, Autovon 226-4405

Principal Investigator: Dr. R. M. Christensen
Lawrence Livermore National Laboratory
Livermore, CA 94550

Objective: Research will be conducted into the mechanics of failure of composite materials, with emphasis on physically-based failure criteria for thick composite laminates.

NONDESTRUCTIVE EVALUATION AND DAMAGE ACCUMULATION OF COMPOSITES
N00014-87-K-0159
April 87 - September 89

Scientific Officer: Dr. Yapa D. S. Rajapakse
Office of Naval Research
Mechanics Division, Code 1132-SM
Arlington, VA 22217-5000
(202) 696-4405, Autovon 226-4405

Principal Investigator: Prof. I. M Daniel
Northwestern University
Department of Civil Engineering
Evanston, IL 60201
(312) 491-5649

Objective: Research will be conducted to understand the process of damage growth in composite laminates subjected to complex loading states and fatigue.

ENVIRONMENTAL EFFECTS AND ENVIRONMENTAL DAMAGE IN COMPOSITES
N00014-82-K-0562
October 84 - September 90

Scientific Officer: Dr. Yapa D. S. Rajapakse
Office of Naval Research
Mechanics Division, Code 1132-SM
Arlington, VA 22217-5000
(202) 696-4405, Autovon 226-4405

Principal Investigator: Prof. Y. Weitsman
Texas A&M University
Department of Civil Engineering
College Station, TX 77843
(713) 845-7512

Objective: Research will be conducted into the effects of stress, temperature and moisture on the mechanical response of polymer composites. Environmentally induced damage growth and its effect on composite response will be investigated.

DYNAMIC MATRIX CRACKING AND DELAMINATION IN COMPOSITE LAMINATES SUBJECTED TO IMPACT LOADING
N00013-84-K-0554
July 84 -May 89

Scientific Officer: Dr. Yapa D. S. Rajapakse
Office of Naval Research
Mechanics Division, Code 1132-SM
(202) 696-4405, Autovon 226-4405

Principal Investigator: Prof. C. T. Sun
Purdue University
School of Aeronautics and Astronautics
West Lafayette, IN 47907
(317) 494-5130

Objective: The propagation of damage in composite laminates due to impact loading conditions will be investigated using theoretical and experimental techniques. Dynamic delamination models will be established. Concepts for controlling impact damage will be explored.

NONLINEAR MODELS FOR BINARY METAL-MATRIX COMPOSITES
N00014-87-K-0176
February 87 - January 89

Scientific Officer: Dr. R. Jones
Office of Naval Research
Mechanics Division, Code 1132-SM
(202) 696-4305, Autovon 226-4305

Principal Investigator: Prof. H. Murakami
University of California, San Diego
Department of Theoretical and Applied Mechanics
LaJolla, CA 92093
(619) 452-3821

Objective: A nonlinear theory for metal-matrix composites will be developed, based on variational principles and multi-variable asymptotic expansion techniques, and accounting for the effects of fiber breakage, fiber-matrix debonding and slip, matrix plasticity and delaminations.

FRACTURE OF METAL-MATRIX COMPOSITES
N00014-85-K-0247
March 85 - May 89

Scientific Officer: Dr. Yapa D. S. Rajapakse
Office of Naval Research
Mechanics Division, Code 1132-SM
Arlington, VA 22217-5000
(202) 696-4405, Autovon 226-4405

Principal Investigator: Prof. G. J. Dvorak
Rensselaer Polytechnic Institute
Department of Civil Engineering
Troy, N.Y. 12181
(518) 276-6943

Objective: Investigations of damage growth and fracture in metal-matrix composites will be conducted using experimental and analytical techniques. The bimodal plasticity theory will be extended to account for matrix hardening and for general mechanical loading states.

MECHANICAL PROPERTIES OF COMPOSITES AT ELEVATED TEMPERATURES
N00014-85-K-0480
July 85 - September 88

Scientific Officer: Dr. Yapa D. S. Rajapakse
Office of Naval Research
Mechanics Division, Code 1132-SM
Arlington, VA 22217-5000
(202) 696-4405, Autovon 226-4405

Principal Investigator: Prof. G. S. Springer
Stanford University
Department of Aeronautics and Astronautics
Stanford, CA 94305
(415) 497-4135

Objective: Mechanics-based models for the temperature dependence of the mechanical properties and failure characteristics of composites will be established.

QUANTITATIVE ULTRASONIC MEASUREMENTS IN COMPOSITES
N00014--85-K-0460
July 85 - September 89

Scientific Officer: Dr. Yapa D. S. Rajapakse
Office of Naval Research
Mechanics Division, Code 1132-SM
Arlington, VA 22217-5000
(202) 696-4405, Autovon 226-4405

Principal Investigator: Prof. W. Sachse
Cornell University
Department of Theoretical and Applied Mechanics
Ithaca, N. Y. 14853
(607) 255-5065

Objective: Research will be conducted to establish quantitative active and passive ultrasonic measurement techniques for characterizing the microstructure and mechanical properties as well as the dynamics of deformation processes in a variety of composite materials.

DAMAGE ASSESSMENT IN COMPOSITES USING ACOUSTO-ULTRASONIC TECHNIQUES
N00014-87-K-0143
February 87 - February 89

Scientific Officer: Dr. Yapa D. S. Rajapakse
Office of Naval Research
Mechanics Division, Code 1132-SM
Arlington, VA 22217-5000
(202) 696-4405, Autovon 226-4405

Principal Investigator: Prof. H. T. Hahn
Pennsylvania State University
Department of Engineering Science and Mechanics
University Park, PA 16802
(814) 863-0997

Objective: Theoretical and experimental studies will be conducted for the application of the acousto-ultrasonic technique for damage assessment in composites.

DYNAMIC BEHAVIOR OF FIBER AND PARTICLE REINFORCED COMPOSITES
N00014-86-K-0280
October 86 - September 90

Scientific Officer: Dr. Yapa D. S. Rajapakse
Office of Naval Research
Mechanics Division, Code 1132-SM

Arlington, VA 22217-5000
(202) 696-4405, Autovon 226-4405

Objective: Research will be conducted into the diffraction of elastic waves by cracks and other inhomogeneities in laminated fiber reinforced composites. Investigations of dynamic material properties of fiber and particle reinforced metal-matrix composites will be conducted.

IMPACT RESPONSE AND QNDE LAYERED COMPOSITES
N00014-87-K-0351
April 87 - December 89

Scientific Officer: Dr. Yapa D. S. Rajapakse
Office of Naval Research
Mechanics Division, Code 1132-SM
Arlington, VA 22217-5000
(202) 696-4405, Autovon 226-4405

Principal Investigator: Prof. A. K. Mal
University of California, Los Angeles
Department of Mechanical, Aerospace and Nuclear Engineering
Los Angeles, CA 90024
(213) 825-5481

Objective: Research will be conducted into wave propagation in composite laminates, with the focus on dynamic loading conditions and theoretical aspects of quantitative acoustic microscopy.

NAVAL AIR DEVELOPMENT CENTER
WARMINSTER, PA 18974-5000
IN-HOUSE

HYBRID COMPOSITE FRACTURE CHARACTERIZATION
September 85 - October 89

Project Engineer: Lee W. Gause
Naval Air Development Center
AVCSTD/6043
Warminster, PA 18974-5000
(215) 441-1330, Autovon 441-1330

Objective: Characterize the strength, mechanical properties, and damage tolerance of woven and hybrid composite structures.

STRUCTURAL DAMPING
October 87 - September 90

Project Engineer: Dr. D. J. Barrett
Naval Air Development Center
AVCSTD/6043
Warminster, PA 18974-5000
(215) 441-1330, Autovon 441-1330

Objective: Improve the damping properties of structures through the redesign of basic structural components as composites of stiffness and damping materials.

ANALYTICAL MODELING OF COMPOSITE INTERFACE MECHANICS
April 88 - September 90

Project Engineer: Lee W. Gause
Naval Air Development Center
AVCSTD/6043
Warminster, PA 18974-5000
(215) 441-1330, Autovon 441-1330

Objective: Understand how interface failure mechanisms develop and influence the properties of resin matrix composites and devise non-linear micromechanics models to describe the behavior at the interface region.

METAL MATRIX CRACK INITIATION/PROPAGATION
September 85 - October 89

Project Engineer: Dr. H. C. Tsai
Naval Air Development Center
AVCSTD/6043
Warminster, PA 18974-5000
(215) 441-2871, Autovon 441-2871

Objective: Characterize the crack initiation/propagation mechanics of silicon carbide/titanium metal matrix composites as applied to landing gear and arrestor hooks in the naval shipboard environment.

CONTRACTS

INFLUENCE OF LOAD FACTORS AND TEST METHODS ON IN-SERVICE RESPONSE OF COMPOSITE MATERIALS AND STRUCTURES
N62269-85-C-0234
June 85 - June 88

Project Engineer: Lee W. Gause
Naval Air Development Center
AVCSTD/6043
Warminster, PA 18974-5000
(215) 441-1330, Autovon 441-1330

Principal Investigator: Prof. K. L. Reifsnider
Virginia Polytechnic and State University
Department of Engineering Science and Mechanics
Blacksburg, VA 24061
(703) 961-5316

Objective: Develop an understanding of the relationship between composite laminate response to high load levels for short time periods and response to low load levels long time periods. Develop an understanding of the relationship between test methods and laminate response. Establish the manner in which these relationships are associated with strength and life. Formulate a mechanistic model which can be used to anticipate long-term behavior.

OUT-OF-PLANE ANALYSIS FOR COMPOSITE STRUCTURES
N62269-87-C-0226
September 87 - March 89

Project Engineer: E. Kautz
Naval Air Development Center
AVCSTD/6043
Warminster, PA 18974-5000
(215) 441-1561, Autovon 441-1561

Principal Investigator: Ken Sanger
McDonnell Aircraft Company
Box 516, St. Louis, MO 63166
(314) 233-9622

Objective: In develop and verify an analysis methodology that provides an up-front capability to identify potential out-of-plane loading situations in composite structures and determine strength and failure mode without resorting to expensive three dimensional finite element analysis.

CERTIFICATION OF COMPOSITE STRUCTURES
N62269-87-C-0259
N62269-87-C-0259.
September 87 - March 89

Project Engineer: E. Kautz
Naval Air Development Center
AVCSTD/6043
Warminster, PA 18974-5000
(215) 441-1561, Autovon 441-1561

Principal Investigator: Ken Sanger
McDonnell Aircraft Company
Box 516, St. Louis, MO 63166

(314) 233-9622
and
Han Pin Kan
Northrop Corporation
Hawthorne, CA 90250
(213) 970-5285

Objective: To expand the certificaiton methodology for composite structures to address adhesively bonded and cocured construction and the effects of in-service impact damage on static strength and fatigue life of composite structures.

DAVID TAYLOR RESEARCH CENTER
BETHESDA, MD 20084-5000
ANNAPOLIS, MD 21842

COMPRESSION RESPONSE OF THICK-SECTION COMPOSITE MATERIALS

Principal Investigator: E. T. Camponeschi, Jr.
David Taylor Research Center, Code 2844
Annapolis, MD 21842
(301) 267-2165, Autovon 281-2165

Objective: Develop an understanding of compression failure for thick section composites.

BEHAVIOR OF COMPOSITES SUBJECTED TO UNDERWATER EXPLOSIVE LOADING

Principal Investigator: Erik Rasmussen
David Taylor Research Center, Code 1720
Bethesda, MD 20084-5000
(301) 227-1656, Autovon 287-1656

Objective: Develop the analytical and experimental techniques required to assess the dynamic capabilities of proposed composite submarine pressure hull structural and material concepts.

COMPOSITE PRESSURE HULL PENETRATION AND JOINT DESIGN

Principal Investigator: M. Brown
David Taylor Research Center, Code 1720.2
Bethesda, MD 20084-5000
(301) 227-1706, Autovon 287-1706

Objective: Develop structurally efficient joint, penetration, and reinforcement concepts for composite pressure hulls; the analytical capability to predict the structural response of these concepts; the experimental capability to verify the validity of the analytical procedures.

COMPOSITE STRUCTURES FOR SURFACE SHIPS

Principal Investigator: M. Critchfield
David Taylor Research Center, Code 1730.2
Bethesda, MD 20084-5000
(301) 227-1769, Autovon 287-1769

Objective: Develop the basic technology to (a) support the applications of composites to naval ship structures including design and analytic methods in structural joints and attachments, and to demonstrate the feasibility of using FRP composites for surface ship structural applications such as deckhouses, stacks and masts, and secondary structures.

NAVAL RESEARCH LABORATORY
WASHINGTON, DC 20375-5000

STRUCTURAL RESPONSE OF DAMAGED COMPOSITES
October 79 - September 88

Principal Investigator: Dr. P. W. Mast
Naval Research Laboratory
Washington, DC 20375-5000

(202) 767-2165, Autovon 297-2165

Objective: Develop a capability for simulating the structural response of composite structures containing a defect or damage.

CONTRACTS

DYNAMIC BEHAVIOR OF COMPOSITES
N00014-86-C-2580
October 86 - September 89

Scientific Officer: Dr. Irvin Wolock
Naval Research Laboratory
Washington, DC 20375-5000
(202) 767-2567, Autovon 297-2567

Principal Investigator: Longin B. Greszczuk
McDonnell Douglas Astronautics Company
5301 Bolsa Avenue
Huntington Beach, CA 92647
(714) 896-3810

Objective: Determine the effect of large area dynamic loading on the mechanical response of composite materials.

AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
MATERIALS LABORATORY

IN-HOUSE

ADVANCED COMPOSITES
WORK UNIT DIRECTIVE (WUD) NUMBER 45
88 October - 89 October

WUD Leader: James M. Whitney
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB OH 45433-6533
(513) 255-9097 Autovon: 785-9097

Objective: The objective of the long term thrust is to develop understanding of deformation and failure process of composite laminates. The short term objectives include the following: (a) development of design methodology of thick composites and their test methods; (b) role of interface in emerging composite systems.

CONTRACTS

IMPROVED TECHNOLOGY FOR ADVANCED COMPOSITE MATERIALS
F33615-87-C-5239
15 Sep 87 - 01 Feb 92

Project Engineer: Marvin Knight
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB OH 45433-6533
(513) 255-7131 Autovon: 785-7131

Principal Investigator: Rebecca C. Schiavone
University of Dayton Research Institute
300 College Park Avenue
Dayton OH 45469

Objective: The objective of this program is to investigate from both an experimental and an analytical standpoint the potential of new and/or modifications of existing matrix materials and reinforcements/product forms for use in advanced composite materials, including processing/mechanical property relationships. Such materials are subsequent candidates for use in advanced aircraft and aerospace structural applications.

MICROMECHANICS OF COMPOSITE FAILURE
F33615-88-C-5420
01 Oct 88 - 30 Sep 92

Project Engineer: Nicholas J. Pagano
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB OH 45433-6533
(513) 255-6762 Autovon: 785-6762

Principal Investigator: Som R. Soni
Adtech Systems Research Inc.
1342 N. Fairfield Road
Dayton OH 45432

Objective: The objective of this program is to provide exploratory development in thermo-mechanical response, model material system development, composite processing, and failure mechanisms investigations of composite and related constituent materials.

DEVELOPMENT OF ULTRA-LIGHTWEIGHT MATERIALS-N
F33615-88-C-5447
29 Apr 88 - 1 Jul 91

Project Engineer: Capt Jocelyn Patterson
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433-6533
(513) 255-9096 Autovon: 785-9096

Principal Investigator: Anne R. Beck
Northrop Corporation
Aircraft Division
One Northrop Avenue
Hawthorne CA 90250

Objective: To demonstrate the potential for advanced ultra-lightweight (ULW) materials and associated processes that will permit a fifty percent reduction in the structural weight of state-of-the-art (SOTA) high-performance aircraft that currently utilize up to ten percent of advanced composite materials in their structures.

DEVELOPMENT OF ULTRA-LIGHTWEIGHT MATERIALS-M
F33615-88-C-5452
13 May 88 - 15 Jul 91

Project Engineer: Capt Jocelyn Patterson
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB OH 45433-6533
(513) 255-9096 Autovon: 785-9096

Principal Investigator: Gail L. Dolan

McDonnell Douglas Corporation
McDonnell Douglas Company
P.O. Box 516
St Louis MO 63166

Objective: To demonstrate the potential for advanced ultra-lightweight (ULW) materials and associated processes that will permit a fifty percent reduction in the structural weight of state-of-the-art (SOTA) high-performance aircraft that currently utilize up to ten percent of advanced composite materials in their structures.

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